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SUMMARY

The unsatisfactory situation regarding the understanding of the stall of swept wings complicates the design of new aircraft. A general hypothesis is presented which serves as a useful guide in determining what must be done empirically to achieve a given set of wing characteristics. Many general and specific studies made to control the stalling of swept wings support the hypothesis; however, it has not been possible to predict quantitatively the wing characteristics.

This state of ignorance regarding swept-wing stall could well be serious. To date the stall control devices in use stem from a background of unswept-wing stalling experience. There is no reason to assume these are necessarily the best solution for the swept wing. A more fundamental understanding of the problem is needed to avoid an unnecessary penalty in low-speed flight performance and safety of swept-wing aircraft.

INTRODUCTION

The increased application of the swept-wing principle to high-speed commercial aircraft has focused attention once again on the difficulties of achieving, with swept wings, sufficiently high maximum lifts together with satisfactory stability and control for landing and take-off. The phrase "once again" is used as a reminder that the problem was faced a decade or more ago with the introduction of swept wings into military aircraft design. The solutions to the high-lift and associated stability and control problems which were adopted for military aircraft cannot necessarily be considered adequate for commercial aircraft. That is, mechanical complication, electronic assistance (in the form of augmentation), and increased approach and landing speeds do not appear desirable for commercial aircraft.

Despite the obvious desirability of achieving a fundamental understanding of these low-speed problems so they could be analyzed in a quantitative sense, it is a fact that most, if not all, of the solutions for the military aircraft were reached in an empirical manner through wind-tunnel studies guided by only qualitative understanding of the phenomena involved. This situation existed not because of lack of interest in the fundamentals of the problem, but simply because time did not allow the painstaking investigations required.

In view of the interest in wider application of swept wings, it is considered of value to review the state of understanding of their low-speed

problems. Obviously, since the information is not complete or definitive, conclusions given are based on a certain amount of conjecture. Nevertheless, it is believed they may serve as a departure point for additional work. The following material is presented with this in mind. The data presented are drawn from many experiments and chosen only to illustrate particular points; no attempt is made to be complete in data presentation; where original data are available, the published sources are cited.

NOTATION

A	aspect ratio
Ae	effective aspect ratio
b	wing span
c	chord
$\bar{c}$	mean aerodynamic chord
C_D	wing drag coefficient
C_L	wing lift coefficient
c_l	airfoil section lift coefficient
C_m	wing pitching-moment coefficient
M	Mach number
P	pressure coefficient
R	Reynolds number
x	chordwise distance from airfoil leading edge
α	angle of attack
λ	taper ratio
Λ	sweep angle
η	local wing spanwise distance, fraction of wing semispan

Subscripts

max maximum

u upper surface

DISCUSSION

The major low-speed aerodynamic problems facing the designer who chooses to use swept wings are the "low" maximum lift and, more important, the appearance, well below maximum lift, of extremely nonlinear pitching-moment curves which usually further limit the "usable" maximum lift. Not surprisingly, potential flow analysis explains none of this although it does, in its various forms, describe with good accuracy all the characteristics of swept wings in the range of low lift coefficients. Since the swept-wing problems at low speeds are a consequence of viscous effects, neglected in potential flow analysis, any improvement in swept-wing characteristics will come from improved understanding and control of the viscous effects. It can be conjectured logically that the viscous effect of major importance to these problems is flow separation related to stall of the straight wing; in the following the term "stalling" will be used to specify C_L values where flow separation appears to have dominant effects on wing aerodynamic parameters.

The first figure, showing results typical of many swept-wing investigations, illustrates the points under discussion. In the low lift-coefficient range the wing characteristics are similar to those predicted by potential flow theory wherein viscous effects are ignored. Above about two-thirds maximum lift, however, the rate of drag rise with lift increases rapidly, the lift curve slope decreases, and the aerodynamic center shifts forward, all apparently results of wing stalling; finally, the measured maximum lift is lower than that which would be anticipated on the basis of experience with unswept wings alone.

Other experimental results, similar to those of figure 1, led to extensive research programs directed at finding some design features which would affect the stalling behavior in a manner to raise the C_L at which stall first occurred, to raise $C_{L_{max}}$, and to avoid the pitch-up associated with forward shift of the aerodynamic center. The solutions were different for each combination of plan-form sweep, aspect ratio, and taper ratio. Many attempts were made to correlate these studies on the basis of geometric parameters; some success was achieved, notably reference 1, but, in general, the correlations were of limited value. It became increasingly clear that some design-chart approach similar to reference 2 was required to provide the designer with a measure of what swept-wing performance might be expected and what geometric factors could be expected to influence this performance.

The success of the method of reference 2 in predicting unswept wing characteristics underscores its basic soundness. Although reference 2 could

not be successfully applied directly to swept wings, it seemed logical to assume this did not invalidate the basic correctness but rather that sweep had introduced new or emphasized hitherto unimportant factors which must be included.

The remainder of this paper, then, will be a discussion of the efforts to refine or extend the principles of reference 2 in an attempt to arrive at an acceptable quantitative understanding of the stalling of swept wings.

Basic Approach to the Prediction of Swept-Wing Characteristics

Prediction of first appearance of stall.- As shown in figure 1, the characteristics of swept wings fall into two regimes: that where the effects of viscosity are small and where it has been demonstrated that inviscid theories apply, and that where the effects of viscosity are dominant. The first step in the study of the stalling of swept wings, then, would be to develop a method that defines adequately the upper limit of the inviscid-flow regime and thus would enable adequate design control of the factors that determine the first appearance of stall.

The method given in NACA TR 572 (ref. 3), with various minor refinements, has been shown to be satisfactory for determining stall on unswept wings. Very important to the usefulness of this method is the degree to which the effects of airfoil section and wing plan form can be studied independently; although such independence cannot be rigorously justified, the benefits from making it a sufficiently accurate approximation are so tremendous that many studies have been directed at reducing the degree of approximation.

At least two changes to the method of TR 572 are necessary to include, correctly, factors known to affect the beginning of swept-wing stall: a span loading theory applicable to the swept wing must be substituted for lifting-line theory, and the concepts of simple-sweep theory must be followed in applying two-dimensional airfoil data. Aside from these changes, the procedure is identical to that of TR 572. As shown in figure 2 for a typical case, the loading theory was used to establish the section lift-coefficient distribution across the wing (shown by the solid curve), and simple-sweep theory concepts were applied to two-dimensional airfoil data to define the distribution of maximum section lift coefficient (shown by the dashed line). The span-loading theory used in place of that based on the lifting line was the one proposed in reference 4. This has been shown, in reference 5, for instance, to be accurate for a wide range of plan forms, but could be supplanted with a still more accurate method. The simple sweep concept was used with two-dimensional airfoil data in order to isolate three-dimensional factors. If instead the streamwise section of a swept wing had been examined (not compatible with "simple-sweep" concepts) the conclusions regarding the three-dimensional factors would differ.

The simple-sweep concept states that the section characteristics on an infinite-span wing do not vary as the wing is yawed, provided the section chosen is normal to the constant percent chord lines and provided the reference velocity chosen is parallel to this section. Included in the "section characteristics" are not only the pressure distributions associated with inviscid flow but also the associated boundary-layer characteristics, whether laminar or turbulent. Thus, the changes in wing characteristics as the infinite wing is yawed are entirely the result of change in reference velocities; for instance, the maximum lift of the yawed infinite wing will be less than that of the unyawed wing exactly in proportion to the square of the ratios of effective to free-stream velocities existing in the case of the yawed wing. What theoretical or experimental proof of the simple sweep concept exists?

The invariance of the pressure distribution has been demonstrated both theoretically (ref. 6) and experimentally. Figure 3 is included to emphasize the point further. Shown on the figure are comparisons of theoretical and measured pressure distributions for airfoil sections taken both parallel to the plane of symmetry and normal to the quarter-chord line of the 45° (ref. 7) and 60° swept wings. The theoretical pressure distributions were obtained through the method of reference 8 as modified in reference 9 for each of the airfoil sections. It can be seen that while the uncambered sections do not show large differences in pressure distribution, these differences occur near the leading edge where, in general, the stalling characteristics are determined. The differences in agreement in the cases of the cambered section are large. This evidence shows that if two-dimensional data are to be used to aid in studying swept-wing stall, they must be applied to a section normal to the quarter chord. The invariance of the laminar boundary-layer characteristics has been shown theoretically in reference 10, and some experimental evidence is included in the same reference. The invariance of the turbulent-boundary-layer characteristics is assumed in order to maintain consistency in the application of the simple-sweep concept. It should be noted that this concept implies that the effective Reynolds number for a section on a swept wing is based on the chord and the component of free-stream velocity normal to the 0.25c line.

The arguments just presented in favor of using the airfoil section normal to the 0.25c line on a swept wing as that one to be related to two-dimensional airfoil characteristics lead to interesting conclusions when the low-aspect-ratio wing of high taper is considered. The limiting case of a triangular wing (swept leading edge, unswept trailing edge) has been examined in an attempt to determine how, if at all, section characteristics could be used. The leading-edge pressure distributions could be related to two-dimensional results through the sweep of the leading edge. On the other hand, pressure distributions over the hinge line of a trailing-edge flap appeared to be relatable to the two-dimensional case through the sweep of the flap hinge line. It is difficult to avoid the conclusion that the simple-sweep concept should be modified to make the reference airfoil in the three-dimensional case a curved one described by lines normal to constant percent chord lines (or, perhaps more accurately, normal to the pressure-distribution isobars). Study of the local stalling behavior of triangular wings encourages

speculation along these lines. It is obvious, however, that this hypothesis would preclude the use of two-dimensional test results. In any event, for high-aspect-ratio wings of moderate taper, it should be possible to avoid the curved airfoil concept. An important exception may be trailing-edge flap effectiveness on plan forms with low sweep of the flap hinge line.

The accuracy of the method under discussion in predicting the first occurrence of stall on swept wings has been examined for a group of wings of widely differing plan form and profile. That is, $c_{l_{max}}$ distributions were determined by means of two-dimensional airfoil data modified by simple sweep concepts, and span loadings were calculated for increasing lift coefficients until that wing lift was determined wherein the span loading curve first reached the $c_{l_{max}}$ curve. The wing variables included sweep, aspect ratio, taper ratio, camber, twist, leading-edge devices of various spans, and trailing-edge flaps. If a sudden increase in the rate of drag rise with lift coefficient is assumed to be the most certain precursor of stall, the results shown in figure 4 are obtained. For these wings, of symmetrical profile and varying sweep and aspect ratio, a degree of conservatism is present in every case - the predicted value is on the average about 20 percent lower than the experimental value. Figure 5 has been prepared to indicate the general order of accuracy of the method in predicting the delay in the first appearance of stall produced by various wing modifications, and leading- and trailing-edge high-lift devices. This figure shows the predicted and measured effects of camber and twist, nose camber, leading-edge slats and flaps, and trailing-edge flaps. The predictions for the modified and/or flapped wings are conservative and to about the same degree as for the unmodified and/or unflapped wings. While the absolute accuracy of the results obtained by application of the method is not outstanding, it is important that the error is always in one direction, and that the effects of design changes are correctly predicted. This is taken as evidence that the procedure is basically correct and accounts for the primary effects of sweep but that secondary, although important, effects have been ignored. The conservatism of the predictions should be emphasized because, as will be discussed later, this is important evidence to be used in developing a hypothesis for the effect of wing sweep on the stalling of airfoil sections.

Prediction of stability changes.- In the foregoing examination of the accuracy of the method, attention has been directed only at the point of sudden drag rise. While this is suitable for evaluating the onset of separation, in practice it is the prediction of more or less sudden pitching-moment changes and their direction which are given prime importance, since stability is thereby directly affected. It has been shown repeatedly that where irregular pitching-moment changes occur, they can be traced to a marked change in section lift-curve slope at some point on the wing span. Since section lift-curve-slope changes generally occur as a result of reaching $c_{l_{max}}$ or being very close to it in the two-dimensional case, it would be expected that the outlined procedure might predict the wing lift coefficient where irregular pitching-moment changes would occur. Further, since the pitching moment of swept wings is largely controlled by the span load distribution (see ref. 11, p. 10), the procedure, in showing the spanwise location of first stall, would be expected to predict the direction of the pitching-moment changes. The

results presented in figure 6 for a representative group of wings show the accuracy with which the predictions can be made. The method predicted correctly that each wing would exhibit a pitch-up moment after the first appearance of stall; it can be inferred, thus, that the method predicted the approximate spanwise location of first stall. In some cases the lift coefficient for pitch-up was higher than that for sudden drag rise; thus, the first appearance of stall does not always produce immediate changes in pitching moment.

The fact that the method appears to predict the spanwise location of stall provides a rational basis for attempting to design wing modifications to force the first appearance of stall far inboard and thus produce pitch-down after first stall. As shown in figure 7, the spanwise variation of $c_{l_{max}}$ for initial stall can be adjusted so that first stall will decrease the wing loading over an area forward of the moment center location and thus produce nose-down moments. Figure 8 shows the effect of such adjustments on the pitching moments of several wings which initially had nose-up moments at high lift. The particular device or devices used to adjust the span loading are indicated for each wing, and in each case the arrangement was supposed to produce nose-down moments at high lift. It can be seen that the prediction was successful in only 50 percent of the cases; this percentage was not increased when a larger number of wings were examined. However, it was noted earlier that the method was most likely to be satisfactory in cases where nose-up moments, or in effect outboard stall, were predicted.

It can be concluded tentatively, then, that the method proposed represents a fundamentally sound approach to the problem of predicting the existence of pitch-up and of prescribing the design changes to delay and possibly to eliminate the pitch-up. However, it must be concluded also that, because of three-dimensional effects, the effectiveness of the stall control device in two-dimensional experiments may not be a measure of its effectiveness on a swept wing.

Effect of Mach number.- All of the foregoing comparisons and remarks have been based on cases where shock-induced stalls were not involved. There was reason to believe, however, that the analysis is applicable to the high Mach number case. Lack of suitably detailed and correlated experimental data (i.e., lack of comparable two- and three-dimensional section data at comparable Reynolds numbers) makes difficult an exact evaluation of the process when applied to high Mach numbers. However, comparisons can be made which serve to encourage further study in this direction. Shown in figure 9 is a correlation of the lift coefficient for sudden drag rise for several wings at two Mach numbers. The comparisons of experiment and prediction are encouraging in spite of the lack of exactly related two-dimensional experimental data. Pitching-moment breaks are compared in figure 10. The lift coefficients for predicted and experimental pitching-moment changes are in fair agreement but the nose-up moment predicted in every case was not always found experimentally and, when it did occur, it was at a higher lift coefficient. Thus the method is conservative at moderate Mach numbers as well as at low Mach numbers.

If the prediction of stall on swept wings is to be extended to high Mach numbers, the existence of an upper limit of Mach number for which the method would be applicable must be recognized. Experimental results indicate that as a Mach number of 1.0 is closely approached, the shock waves emanating from the wing-fuselage intersection and from the wing tip exert a controlling effect on the stalling pattern of the wing; under these conditions any attempt to apply reasoning based on two-dimensional concepts is obviously illogical.

The Importance of Three-Dimensional Viscous Effects in the Design of Swept Wings

In order to illustrate and establish some quantitative measure of the magnitude of the three-dimensional effects on stalling, reference is made to material in a report comparing the two-dimensional characteristics of an airfoil section with those for the same section on a swept wing (ref. 12). The comparisons presented in that report are illustrated in figure 11 for a 45° sweptback wing having an airfoil section for which detailed two-dimensional data exist. Adjusting the two-dimensional section lift-curve slopes to correspond to those given by Weissinger theory for several span stations enables a direct comparison to be made with data obtained experimentally at each station on the three-dimensional wing. In these and all following comparisons, the section characteristics for the swept-wing sections are for a section normal to the quarter-chord line of the wing and are based on velocity parallel to this section. A most striking point is that at all stations except the most outboard, the two-dimensional section maximum lift is definitely exceeded and in increasing measure for further inboard stations. These data imply, then, that at no place on the span are the three-dimensional effects (probably characterized by the spanwise boundary-layer flow) detrimental to $c_{l_{max}}$, and at most places are favorable. This spanwise flow should, then, be considered a strong, natural form of boundary-layer control.

The phenomenon just described is not unique to the wing in question. Similar results are shown in figure 12 for a group of wings typical of almost all wings for which such comparisons can be made. From these data it can also be seen that not only is maximum lift always increased toward the inboard stations, but the percent increment is increased as sweep is increased. For the more highly swept wings, it becomes impossible to determine a $c_{l_{max}}$ for the inboard stations. The existence of this phenomenon explains the conservatism of the method previously discussed, since that method ignored any such increase in inboard $c_{l_{max}}$ values.

As a first step in the process of accounting quantitatively for the existence of this effect in swept-wing design, it is necessary to determine just how this natural boundary-layer control is applied. Such an understanding can come from examination of the form of the separation which limits $c_{l_{max}}$ for the two-dimensional and three-dimensional cases. Before doing this, it is desirable to clarify what is meant by "form of separation."

The pattern of separation existing just prior to the maximum lift of an airfoil section has three general forms, shown in figure 13. First is that common to thick or highly cambered sections on which separation first appears at the trailing edge, then spreads slowly forward with increasing angle of attack to finally fix $c_{l_{max}}$; the related pressure distributions show a distinct and sharp peak at the leading edge, a lack of complete pressure recovery at the trailing edge, and an area of constant pressure coefficient over the aft portion where separation exists. The second pattern is that common to very thin sections on which separation of flow at the leading edge appears, followed by reattachment of flow farther aft, and where the point of reattachment moves aft with increasing angle of attack to finally fix $c_{l_{max}}$ as it reaches the trailing edge; the related pressure distribution shows a slight peak at the leading edge followed by a region of relatively constant pressure aft to the point of reattachment, and then recovery to essentially free-stream pressure. The third pattern is that common to sections of about 10-percent thickness and little camber on which both types of separation appear and for which $c_{l_{max}}$ is fixed when the forward-spreading trailing-edge separation becomes sufficiently extensive or reaches the aft-moving point of reattachment of the leading-edge separation; the related pressure distribution shows both a loss of the sharp peak at the leading edge and lack of recovery at the trailing edge, with some evidence of pressure recovery between these points. On the basis of these distinctions and from examination of the chordwise pressure distributions just prior to stall of a given airfoil section in two- and three-dimensional flow, an insight can be had into the mechanism of the natural boundary-layer control on swept wings.

Consider again the case of this previously mentioned 45° sweptback wing. Shown in figure 14 is a comparison of the two-dimensional pressure distribution just prior to $c_{l_{max}}$ with the corresponding ones for the various spanwise sections of the wing. Two-dimensional pressure distributions show the typical evidence of both leading- and trailing-edge types of separation - both a loss of the sharp leading-edge peak and a lack of recovery at the trailing edge. Pressure distributions for several stations on the span of the swept wing indicate the same type of separation pattern over the outboard part of the span, but a change to the thin airfoil, leading-edge type of separation on the inboard sections. From this, it can be judged that the boundary-layer control is increasingly effective for the trailing-edge type of separation as the stations are nearer the root. Now consider the same wing swept to 60°. As indicated in figure 15(a), the separation pattern has been changed from that for the wing at 45° sweep; across the entire span of the 60° swept wing the sections show only the leading-edge separation just prior to section maximum lift. It appears then that increasing sweep intensifies the boundary-layer control at the trailing edge. Now examine the results shown in figure 15(b) for the 45° swept wing with a highly cambered section, an NACA 64A810. The wing is also twisted, but since this adjusts only span loading, it should not significantly disturb the balance between leading- and trailing-edge boundary-layer control. As indicated by the pressure distributions in figure 15(b), two-dimensional tests show the section to have extensive trailing-edge separation just prior to maximum lift. Note the constant value of pressure coefficient over the rear 25 to 30 percent of the section. Data obtained from various stations on the wing show the trailing-edge

separation to be almost entirely suppressed at all but the outermost stations on the wing. The two cases just discussed cover two types of section stall which are altered when under the influence of the natural boundary-layer control existing on a swept wing. Again, these should not be looked upon as unique examples, but rather as typical of what has been found to occur in other cases. The effect of wing sweep on the third type of section stall, that originating wholly from the leading edge, cannot be ascertained because of the lack of comparable data. However, it might be inferred that such leading-edge separation will be delayed also, increasingly so with sweep or inboard location, since air-flow studies show a strong spanwise flow of the boundary layer along the leading edge as well as aft on the wing.

The two major effects of wing sweep, suppression of inboard stall, particularly at the trailing edge, through the natural boundary-layer control just discussed, together with the outboard movement of the peak of the span loading distribution, which is increased as taper is increased, combine to produce a stalling pattern which is unlike any commonly experienced by unswept wings.

It has been shown (ref. 13) that when a thin airfoil section is at appreciable angle of attack, but below maximum lift, the area of separation lying near the leading edge contains a strong vortex; as the angle of attack is increased, the rearward edge of the area of separation moves toward the trailing edge of the section, and the enclosed vortex increases in size and strength, becoming quite apparent before the separation spreads to the trailing edge and $c_{l_{max}}$ is reached. On a swept wing the natural trailing-edge boundary-layer control in delaying normal stall causes this phenomenon to appear on sections of much greater thickness than on unswept wings. Also, because of the usual section lift-coefficient distribution on a swept wing, the vortex appears, first, at the tip and spreads slowly toward the root as wing angle is increased. In many cases, before the leading-edge vortex spreads to the root, the tip sections have complete separation, and the vortex has curved back to leave the wing at the farthest inboard point where separation has reached the trailing edge. As angle of attack is increased, both the origin of the vortex and the point at which it leaves the wing move inboard. This inboard movement of the tip vortex is particularly serious, for it produces much of the drag at high lift (since it effectively reduces the wing aspect ratio) and many of the stability difficulties encountered where a high-placed horizontal tail is used (since it causes rapid increases in downwash in the plane of symmetry).

The foregoing analysis, even though largely qualitative, offers an explanation for many of the observed characteristics of swept wings and enables rational speculation as to the best way to improve swept-wing characteristics and as to probable limits of improvement. No attempt will be made here to explore in detail all the implications for all of the wing characteristics; pitching moment will be given primary attention.

The effect of wing sweep on section stalling limits the direct application of section data proposed in the method outlined earlier. In particular, it is clear why the method failed when it was used to adjust section $c_{l_{max}}$ to force inboard stall to occur first, and thus give nose-down moments.

Because the maximum lift of the inboard sections is far above the two-dimensional values, it is not possible from two-dimensional considerations alone to know when the $c_{l_{max}}$ of outboard sections is sufficiently increased; this increase must not only exceed that of the inboard sections, but by sufficient margin that outboard stall will not be precipitated by flow of air from the stalled inboard area. To demonstrate the powerful effect of sweep on this problem, three wings of different sweep, 35°, 45°, and 60°, will be considered. Each wing when unmodified showed first stall at the tip and resulting nose-up moments. It was apparent that if a stalled area could be initially produced anywhere inboard of the tip, nose-up moments would be reduced. By means of leading-edge slats, increases (based on two-dimensional considerations) were made in the $c_{l_{max}}$ values of sections lying within various percent spans of the outboard portion of each wing. As shown in figure 16, inboard stall and nose-down moments were produced in the case of the 35° swept wing where section $c_{l_{max}}$ values were increased over the outboard 40, 61, and 75 percent of the span. Results for the 45° wing are given in figure 17. Note that the initial point of stall could be moved into the 60- or 40-percent span point, although only the latter produced the desired nose-down moments. In contrast to the 35° swept wing, when it was attempted to move the initial stall on the 45° swept wing into the 20-percent span point, it was found impossible, as initial stall again appeared at the tip. Results presented in figure 18 for the 60° wing show that inboard stall and nose-down moments could not be produced in this case. It is clear that as sweep was increased, the natural boundary-layer control increased the inboard section maximum lift to a point where it roughly equalled that of the slotted outboard sections, and the effect of the discontinuity in spanwise distribution of $c_{l_{max}}$ was lost. Note that for the 60° swept wing this was true even as far outboard as the 60-percent span point, in contrast to the 45° swept wing. (See also ref. 13.)

Although the foregoing shows what is probably the most important three-dimensional effect of sweep not considered by the simple analysis first presented, there is a second important factor to be considered. As demonstrated earlier, wing sweep has also the effect of changing the location of airfoil-section separation from the trailing edge to the leading edge, with the effect becoming stronger toward the root. This effect must also be considered when the effect of separation-controlling devices is estimated from two-dimensional data. For example, consider the effect of a leading-edge slat on a wing swept 45° and then 60° (fig. 19). The basic airfoil was again a 64A010 which, as noted earlier, has two-dimensional separation both at the leading and trailing edges just prior to maximum lift. A slat, if properly drooped, delays primarily the appearance of leading-edge separation on a two-dimensional airfoil. As figure 19 shows, the slat also served this purpose near the tip of a 45° swept wing; thus prior to $c_{l_{max}}$ the section pressure distribution shows a loss of pressure recovery at the trailing edge, indicating stall is initiated by trailing-edge separation. On the contrary, the same slat on the same wing swept to 60° could not contain the leading-edge separation; just prior to $c_{l_{max}}$ the section pressure distribution shows a loss of leading-edge pressures, while full pressure recovery is realized at the trailing edge. It is inferred that the natural boundary-layer control was

more powerful in containing trailing-edge separation than was the slat in containing leading-edge separation despite the slat effects found in two-dimensional studies. Another wing, swept 63° and with an airfoil section very similar to a 64A010, was equipped with area suction boundary-layer control at the leading edge. It was possible, with this form of boundary-layer control, to prevent leading-edge separation from preceding trailing-edge separation, as shown on the pressure distribution on the right of figure 19; note that a very high leading-edge peak is reached before pressure recovery at the trailing edge decreases. It is interesting to note that this condition was reached at a section lift coefficient nearly twice that reached on the tip of the 60° swept wing where leading-edge separation was already evident. In attempting to control outboard wing stall, consideration must be given not only to the section $c_{l_{max}}$ that must be achieved, but also to the fact that the two-dimensional stall pattern may be shifted to make leading-edge stall the dominant problem.

This examination of the two factors which appear to affect most significantly the problem of properly controlling wing stall, makes it possible to consider the general case and show probable reasons for the success or failure of some of the stall-controlling devices which have been tried. Obviously, the most desirable solution is to increase the maximum lift of outboard sections sufficiently, since this also increases total maximum wing lift. However, it appears from examination of inboard $c_{l_{max}}$ that for wings swept more than 45°, the natural boundary-layer control causes inboard $c_{l_{max}}$ values that will be difficult or impossible to exceed at the tip no matter what device is used to increase $c_{l_{max}}$; for instance, on a typical wing of 45° sweep, it would be necessary to exceed two-dimensional maximum section lift coefficients of 2.8 on the outboard stations, whereas on a typical wing of 60° sweep, an outboard $c_{l_{max}}$ of over 3.2 (two-dimensional) would be required. It has been found that on a 63° swept wing, two-dimensional lift coefficients of about 3.9 were realized at the tip without successfully moving stall inboard. Where this approach becomes impossible, the alternative of reducing inboard $c_{l_{max}}$ must be resorted to even at the cost of reducing wing $C_{L_{max}}$. Two general approaches are possible: first, to spoil the flow over inboard sections and thus counteract the effect of boundary-layer control and, second, to minimize to the degree necessary the boundary-layer control at the inboard stations. The first approach does not appear promising, although studies are so limited that a definite conclusion is not possible. For instance, leading-edge spoilers were attached to the inboard leading edge of the 35° sweptback wing, the results for which are shown in figure 20(a). The spoilers were of a size that has been shown by two-dimensional tests to reduce markedly maximum lift. The measured pitching moments show no evidence of the nose-down tendency, which would accompany root stall, for any of the spoiler spans tested, although there is evidence that the root disturbance slightly reduced the maximum lift of the tip sections. Tuft studies showed the spoiler action to be confined to an area just aft of the spoiler and, in opposition to two-dimensional experience, showed complete reattachment of flow over the rearward area. It would appear, then, that for even 35° of sweep the boundary-layer control is sufficient to overcome conventional spoiler action on inboard sections; hence, wings of greater sweep cannot be given nose-down moments in this manner (also see ref. 14).

A more promising manner of obtaining nose-down moments (although still at a cost of reducing maximum wing lift) is to minimize the boundary-layer control on inboard sections. As an example, consider the 35° swept wing just discussed with a small discontinuity added to the wing leading edge at the 20-percent span point (see fig. 20(b)). Tuft studies showed the effect of such a device was to create a vortex lying just above the surface of the wing and rotating so as to sweep the boundary layer inboard, thus minimizing the outboard drain. Under these conditions, the root area stalled sufficiently early to provide the nose-down moments. Similar effects have been noted in the case of partial-span leading-edge devices which were able to give nose-down moments. The effectiveness of such devices has been found to be measurably reduced when the inboard end was faired smoothly to eliminate any sharp discontinuity.

Perhaps a more direct way of minimizing the boundary-layer control on inboard sections is through the use of physical dams or fences to stop or reduce the spanwise boundary-layer control. Experience has shown that only under certain conditions can a fence prove successful; application of the reasoning of this paper shows the factors which should govern successful action of fences. At most a fence should cause the sections just inboard of it to have two-dimensional maximum lift and type of stall, whereas the sections just outboard should show all the effects of the natural boundary-layer control. If advantage is to be taken of this to produce nose-down moments at high lift, further steps must be taken. Thus, for the case of constant sections, the wing span-load distribution must be adjusted by plan form or twist to give a maximum loading where first stall is desired; if wing section alone is varied, then the maximum lift of the sections outboard of that one where first stall is desired must be sufficiently higher than inboard sections to sustain the additional load introduced by sweep, taper ratio, and/or aspect ratio. When a proper relation is attained between section loading and $c_{l_{max}}$, then the location of the fence must be considered. If the section stalls two-dimensionally from the trailing edge, then the fence must be placed aft to stop boundary-layer control at that point. Under any circumstances, it is not likely that a fence will have a dominant effect, but can only be of aid in obtaining full benefit from other devices. With this reasoning in mind, it is useful to examine several cases where fences have been tried.

It has been implied that on the thin swept wing with symmetrical sections, a fence is likely to prove ineffective. Figure 21 shows the reason for this. It is evident that even if the fence wholly overcame the boundary-layer control, inboard stall would not result. Figure 22 shows a typical case in which such is the result. For wings of little sweep - probably 35° or less - where the span loading is not appreciably different from the unswept wing and where the boundary-layer control is not strong, it is possible a fence could prove effective.

A number of cases can be shown where a fence was able to increase the effectiveness of a partial-span leading-edge device. This effectiveness varies in degree from simply increasing the nose-down tendency near maximum lift to producing a nose-down moment where nose-up moments existed without the fence. Generally, fences become most necessary as sweep increases, but it is

also evident their effectiveness vanishes with sufficient sweep. It is evident the action of a combination of partial-span leading-edge device and fence is very similar to that of the leading-edge device alone, where an aerodynamic fence in the form of a vortex has been shown to exist. Typical results are shown in figure 23(a) (reproduced from fig. 18 of ref. 14) and figure 23(b) (reproduced from fig. 7 of ref. 15). In the first case of lesser sweep, existing nose-down moments were increased, and in the second case of greater sweep, nose-up moments were nearly eliminated.

It is also shown, in the references just quoted, for example, that the optimum combination of fence and leading edge varies with trailing-edge flap deflection. That such should be the case is clear when consideration is given the changes in span-load distribution and spanwise section maximum lift distribution engendered by flap deflections.

The foregoing discussion is directed only at demonstrating the probable action of fences on swept wings. There are many details regarding fences which, in all likelihood, will never be subject to generalization since they, in turn, are affected by each variable in the wing's geometry. Thus, the exact values of fence location, spanwise and chordwise, and fence height and chordwise extent for maximum fence effectiveness must undoubtedly be found experimentally for each combination of wing plan form, including leading-edge and trailing-edge devices and airfoil sections. It is believed, however, that consideration of the principles discussed will aid in directing such research.

Reynolds number effects.- All of the reasoning and conclusions drawn to this point have been based on results obtained at high Reynolds number. It is of interest, and particularly with regard to the action of fences, to consider the effect of reduced Reynolds number on swept-wing characteristics.

The argument has been advanced and supporting evidence produced that the effective velocity in the case of the airfoil section on a swept wing is closely that one normal to the wing quarter-chord line. Similar arguments can be advanced, although the supporting evidence is meager, that the effective Reynolds number should also be based on the effective velocity and the chord normal to the quarter-chord line. If this is so, then it is apparent that the effective Reynolds number of any airfoil section on the swept wing is less than the Reynolds number based on the MAC by a factor equal to, on the average, the cosine squared of the angle of sweep. Thus, for a wing of 45° of sweep, the Reynolds number based on the MAC must be over 2×10^{6}, to reach a section Reynolds number of 1×10^{6}. In small-scale tests, then, section Reynolds number can become extremely low. Reference 16 shows that the characteristics of airfoil sections, particularly the values of maximum lift, undergo marked changes in the low Reynolds number range. It would be expected, as has been shown, that swept wings would be excessively sensitive to Reynolds number effects even over a Reynolds number range where straight wings snow only minor effects.

The effects of Reynolds number on swept wings are further complicated by the spanwise boundary-layer flow. For example, as reference 16 indicates, the effect of very low Reynolds number is to promote extensive trailing-edge

separation at low angles of attack; this, of course, is the very effect the boundary-layer drain tends to overcome. Thus, the boundary-layer drain can be considered to increase effectively the Reynolds number of inboard sections; in this way the effective Reynolds number range encompassed by airfoil sections on a swept wing may include that wherein there is a great change in section characteristics. The maximum lift would not be expected to reflect this effect, since for both large-scale and small-scale swept wings this occurs after a large part of the tip is stalled and since the effect of Reynolds number on lift of a stalled surface is small, the region of great differences in characteristics due to Reynolds number has disappeared. However, the pitching moments at higher lift, in particular, would show large Reynolds number effects, since as previously noted, the section maximum lift coefficients dominate these characteristics. Figure 24 is typical of such results.

While insufficient data exist to document thoroughly these Reynolds number effects, the effect of Reynolds number on the action of fences can be interpreted as a verification of the existence of these effects. It has been proposed earlier that a fence, to a large degree, acts simply as an additional wing root in that it increases the boundary-layer control just outboard of it, and, of course, reduces that just inboard. Thus, at low Reynolds number a fence can effect a very large change in the maximum lift of sections on either side of it. As Reynolds number is increased, the change in maximum lift, and, accordingly, the effectiveness of the fence, becomes much less, so much so in some cases that the fence will control the stall in tests at low Reynolds number but not at high Reynolds number.

Care must be taken also that Reynolds number effects do not obscure the effectiveness of fundamental design parameters. For example, using both camber to increase section $c_{l_{max}}$ and twist to adjust span loading would seem pertinent to swept-wing design. Figure 25 shows, however, that the apparent usefulness of these design parameters would be very different, depending on the Reynolds number of the experimental work.

It is recognized that these comments regarding the effect of Reynolds number on the characteristics of swept wings are only qualitative. It must be remembered, however, that the effect of Reynolds number on the maximum lift of two-dimensional airfoil sections is "understood" quantitatively only to the extent that a vast amount of experimental data has been used to arrive at some empirical factors. No such collection of data exists for the far more complex case of the swept wing. It is probable that Reynolds number effects for swept wings are far different from those for straight wings. Thus, any attempt to predict Reynolds number effects on swept wings which is based wholly on unswept-wing experience must be considered highly suspect. A basic consideration for swept wings is the effect on wing stall of the spanwise flow of the boundary layer.

State-of-the-Art Summary

The foregoing discussion enables a state-of-the-art summary of the current understanding of the stalling of swept wings. The salient points can be stated as follows:

(a) Inviscid flow theories which are a modified form of the analysis of TR 572 conservatively predict the first appearance of stall on a swept wing.

(b) Up to the first appearance of stall, a reference airfoil on the swept wing chosen normal to the quarter-chord line of the wing generally permits reasonable comparisons between two- and three-dimensional pressure distributions.

(c) The conservatism cited in (a) above is a consequence of a spanwise flow of the boundary layer which acts as a natural boundary-layer control system and increases section maximum lift on the swept wing above two-dimensional values.

(d) Once local stall has appeared, the spanwise boundary-layer flow serves to change the stalling characteristics of the unstalled sections so they have little resemblance to two-dimensional results, either in the value of the lift coefficient at which stall occurs or in the type of stall demonstrated.

(e) Stall control devices on a swept wing are important in affecting local section lift and the spanwise boundary-layer flow.

With this summary in mind it is possible to examine the problem of developing a procedure to predict swept-wing stalling characteristics with at least the accuracy demonstrated by TR 572 for unswept wings. It must be recognized that the success of TR 572 depends to a very large degree on the fact that experimental two-dimensional section data were used to produce satisfactory answers. This intuitively logical step cannot be employed for swept wings because three-dimensional boundary-layer conditions on a swept wing differ so from any boundary-layer conditions on a two-dimensional airfoil that stalling behaviors are unrelated. Several detailed studies of boundary layers on swept wings failed to uncover any relation, rigorous or empirical, between two- and three-dimensional boundary layers which would aid in understanding or predicting three-dimensional separation. The difficulties encountered in attempts to prescribe theoretically the energy transfer, or shearing stress, in the two-dimensional turbulent boundary layer indicates that there is little possibility of realizing success with fundamental studies of three-dimensional boundary layers.

Ames Research Center
National Aeronautics and Space Administration
Moffett Field, Calif., April 16, 1964

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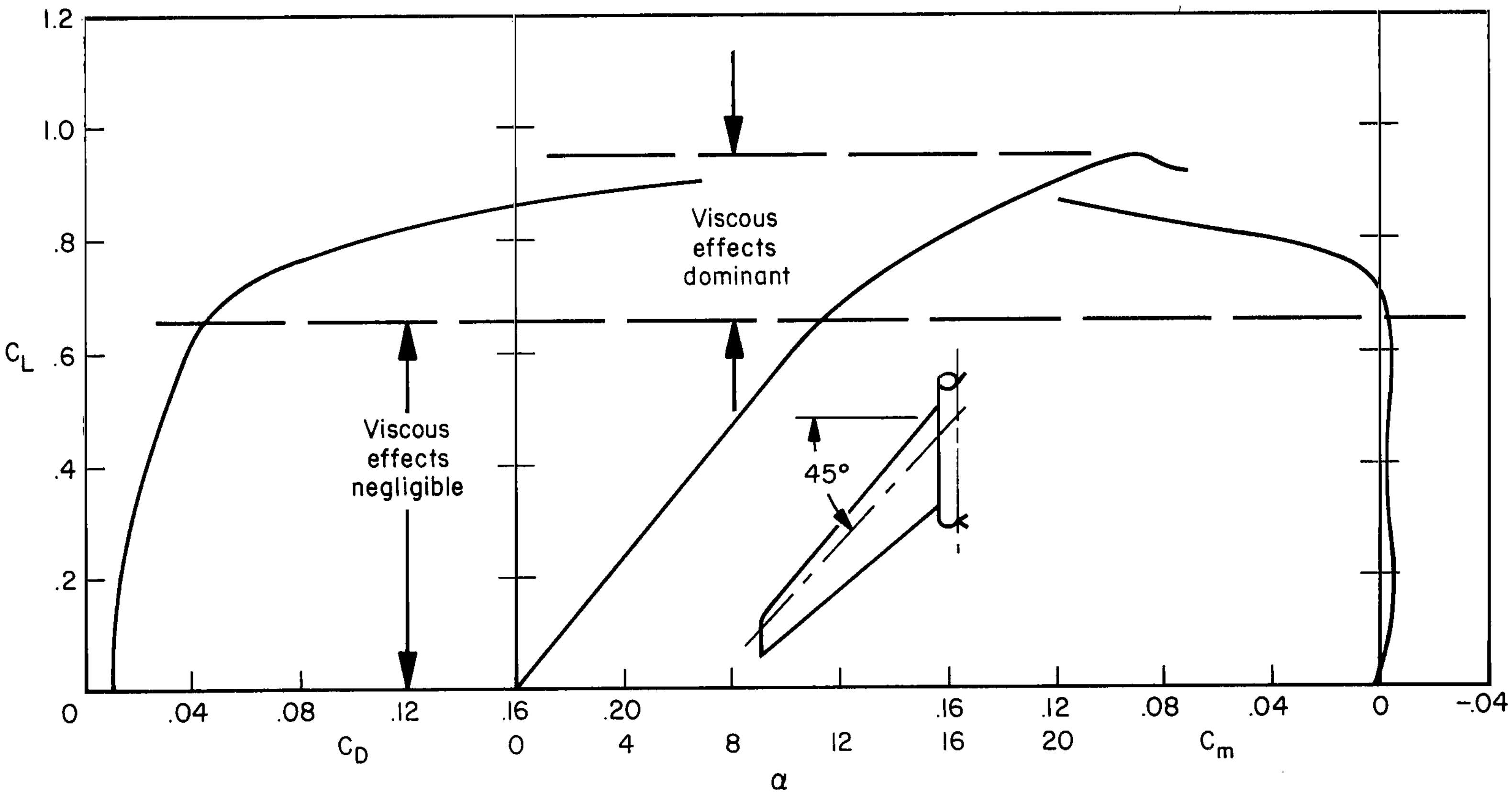


Figure 1.- Aerodynamic characteristics of a typical swept-wing configuration.

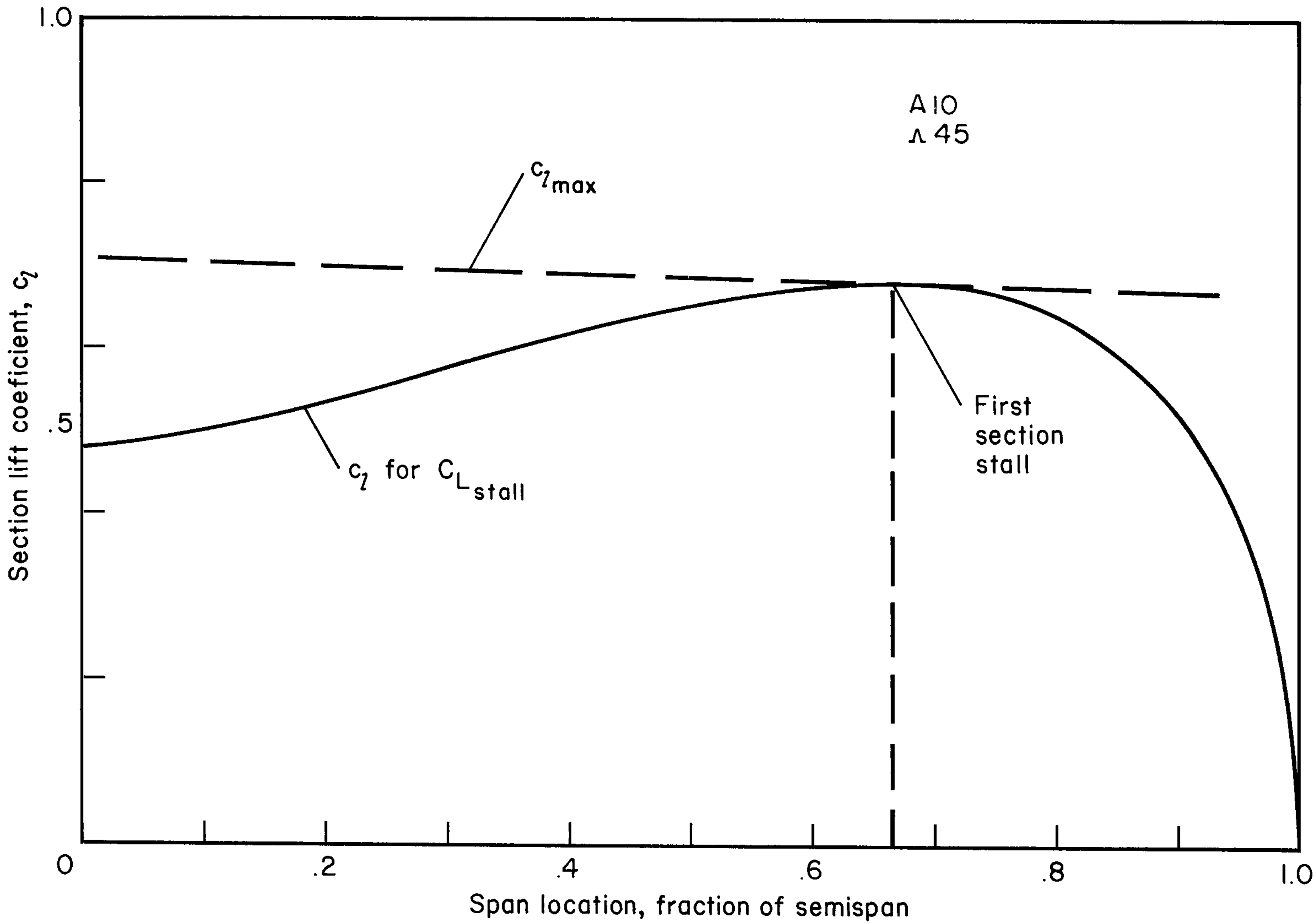
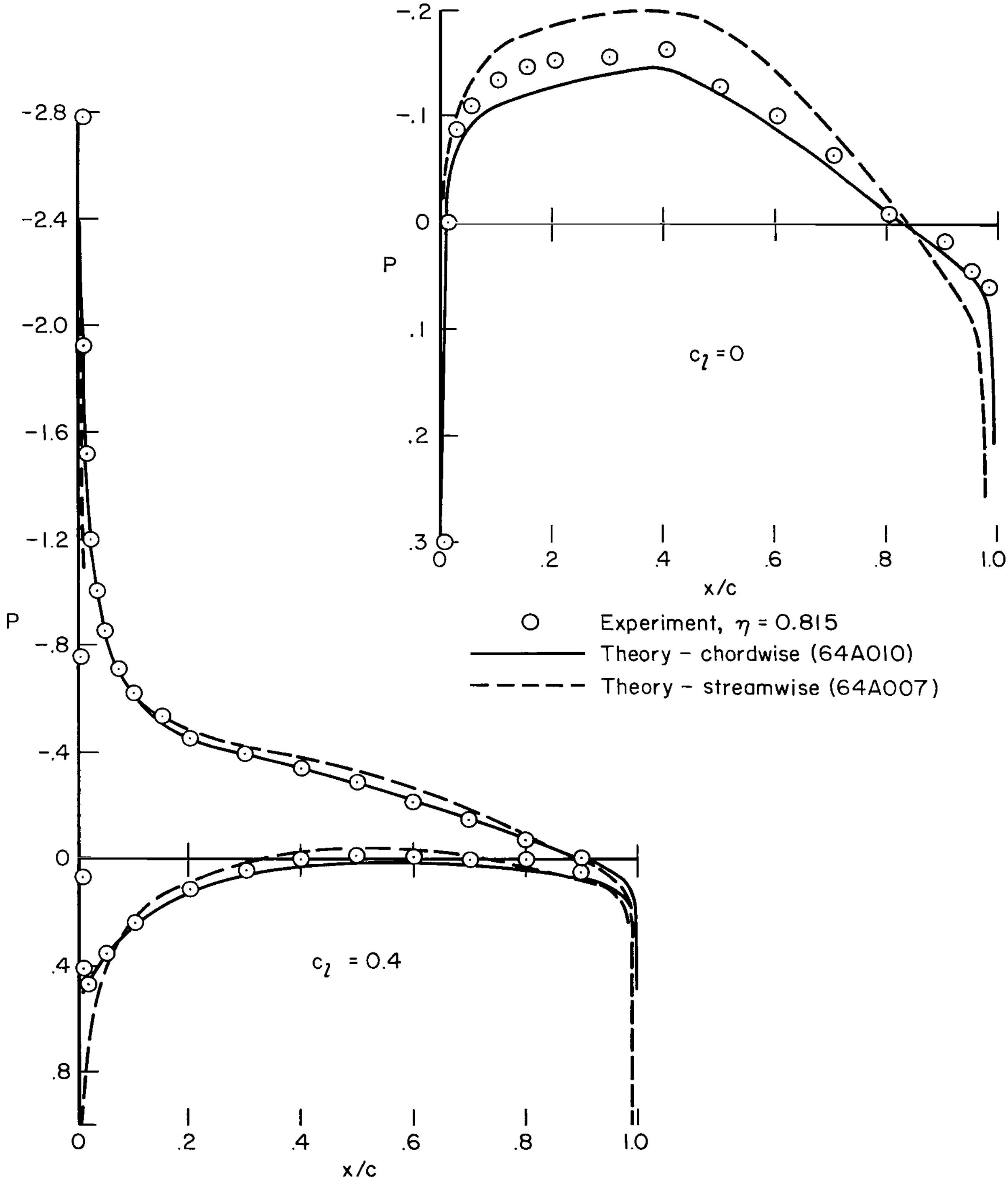
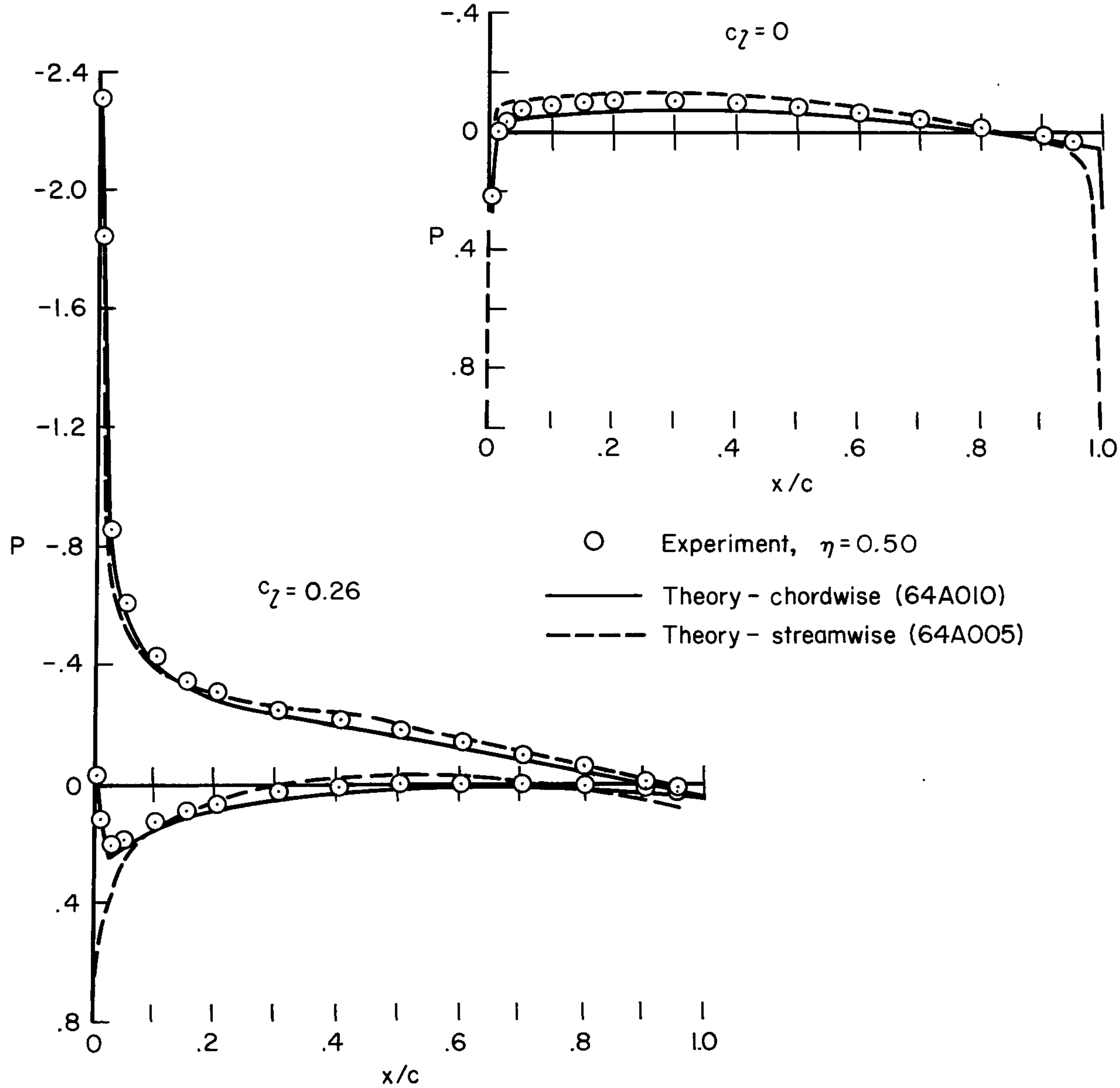


Figure 2.- Determination of first section stall on a swept wing.



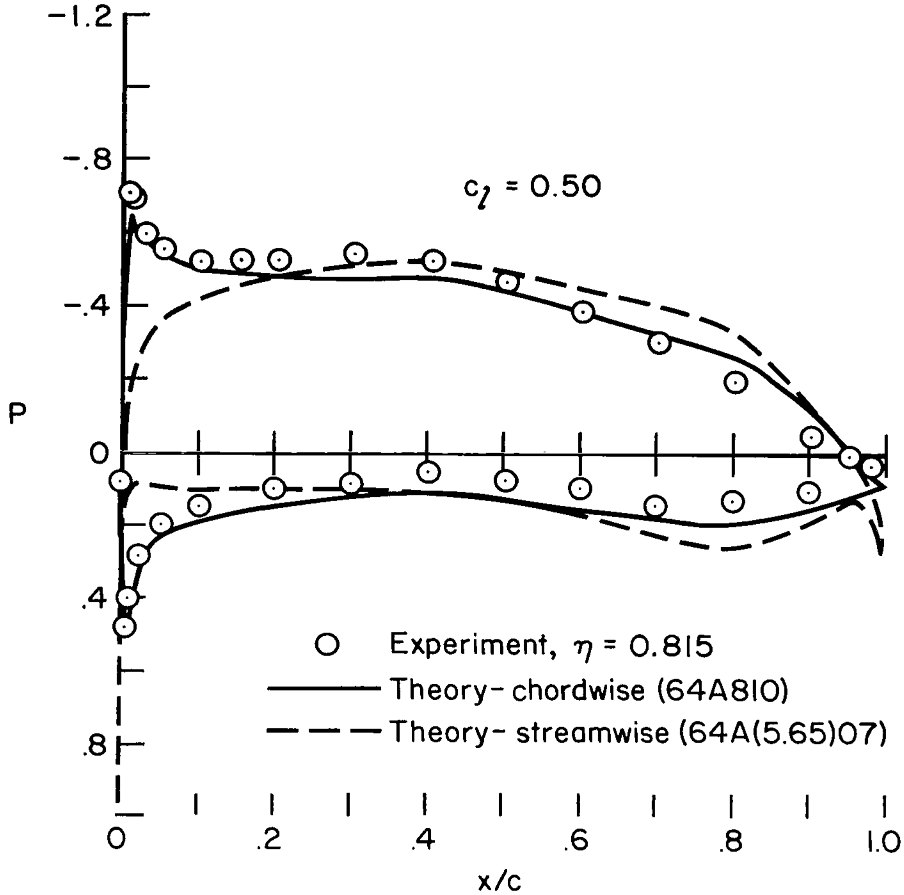
(a) 45° swept wing with NACA 64A010 sections normal to the quarter chord line.

Figure 3.- Comparisons of theoretical section pressure distributions with experimental loadings on finite wing panels.



(b) 60° swept wing with NACA 64A010 sections
normal to the quarter-chord line.

Figure 3.- Continued.



(c) 45° swept wing with NACA 64A810 sections normal to the quarter-chord line.

Figure 3.- Concluded.

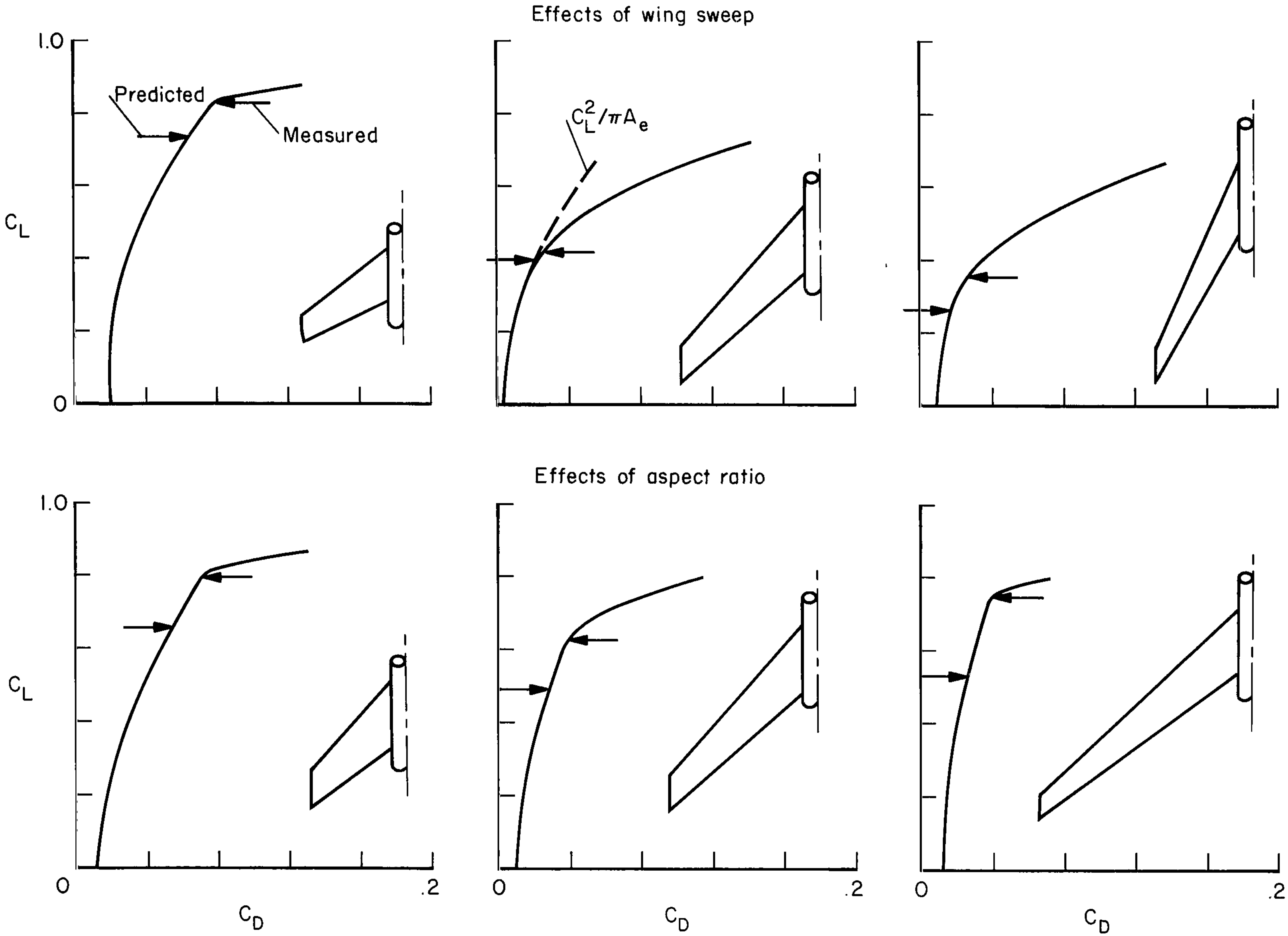


Figure 4.- Drag characteristics of several wings used to indicate first section stall.

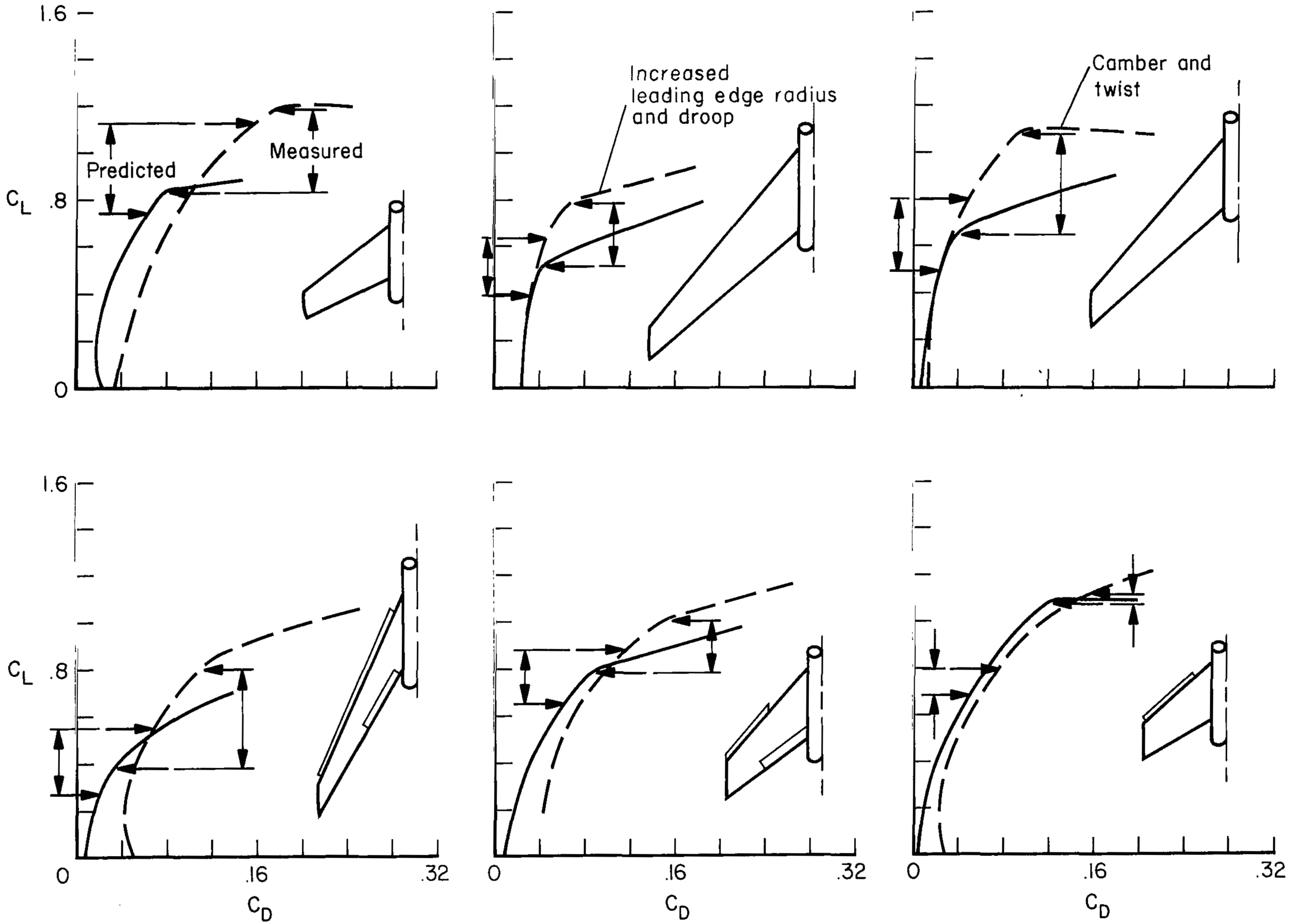


Figure 5.- Comparison of predicted and measured values of wing lift coefficients for drag rise on modified wings.

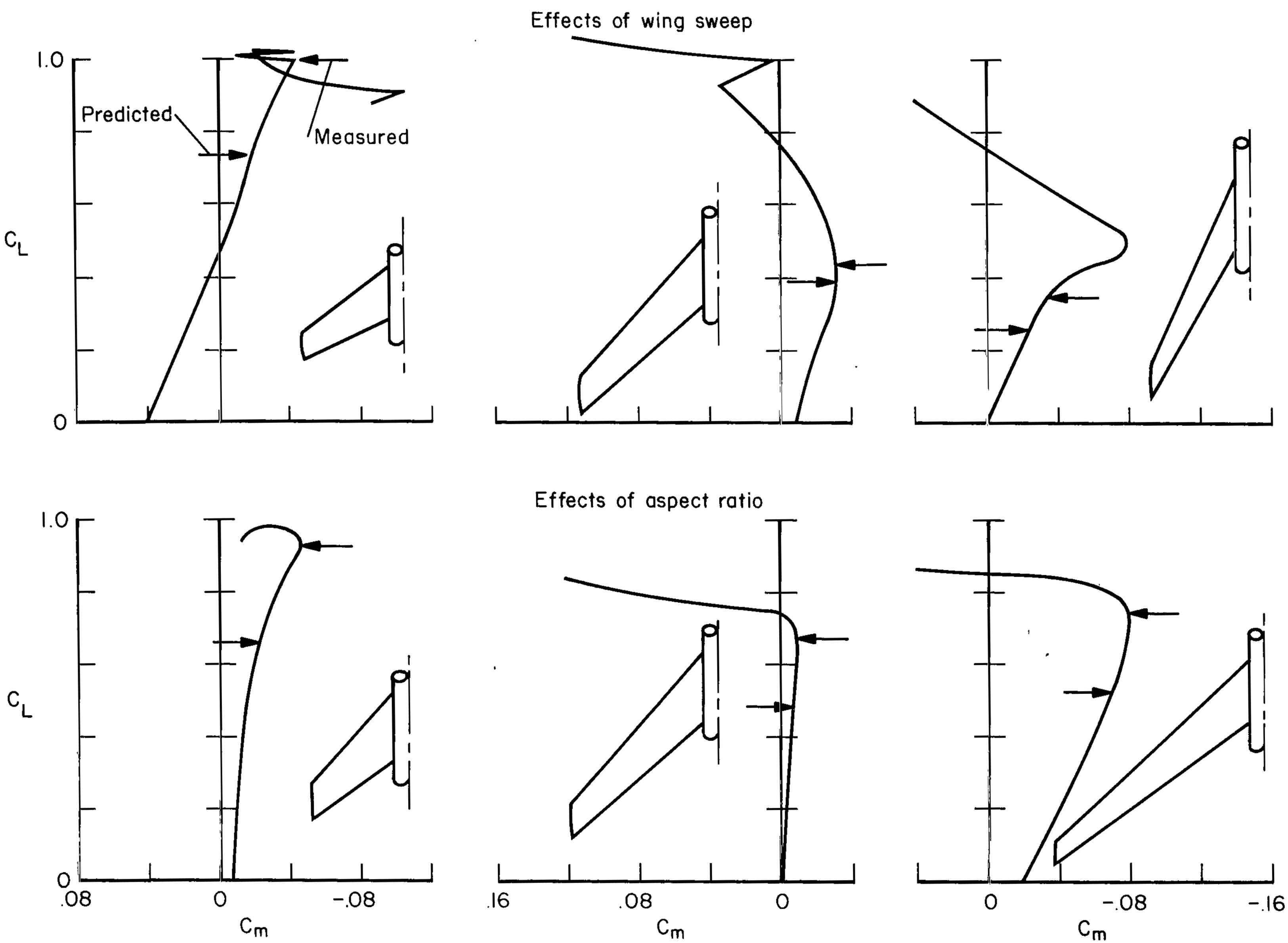


Figure 6.- Pitching-moment characteristics used to indicate first section stall.

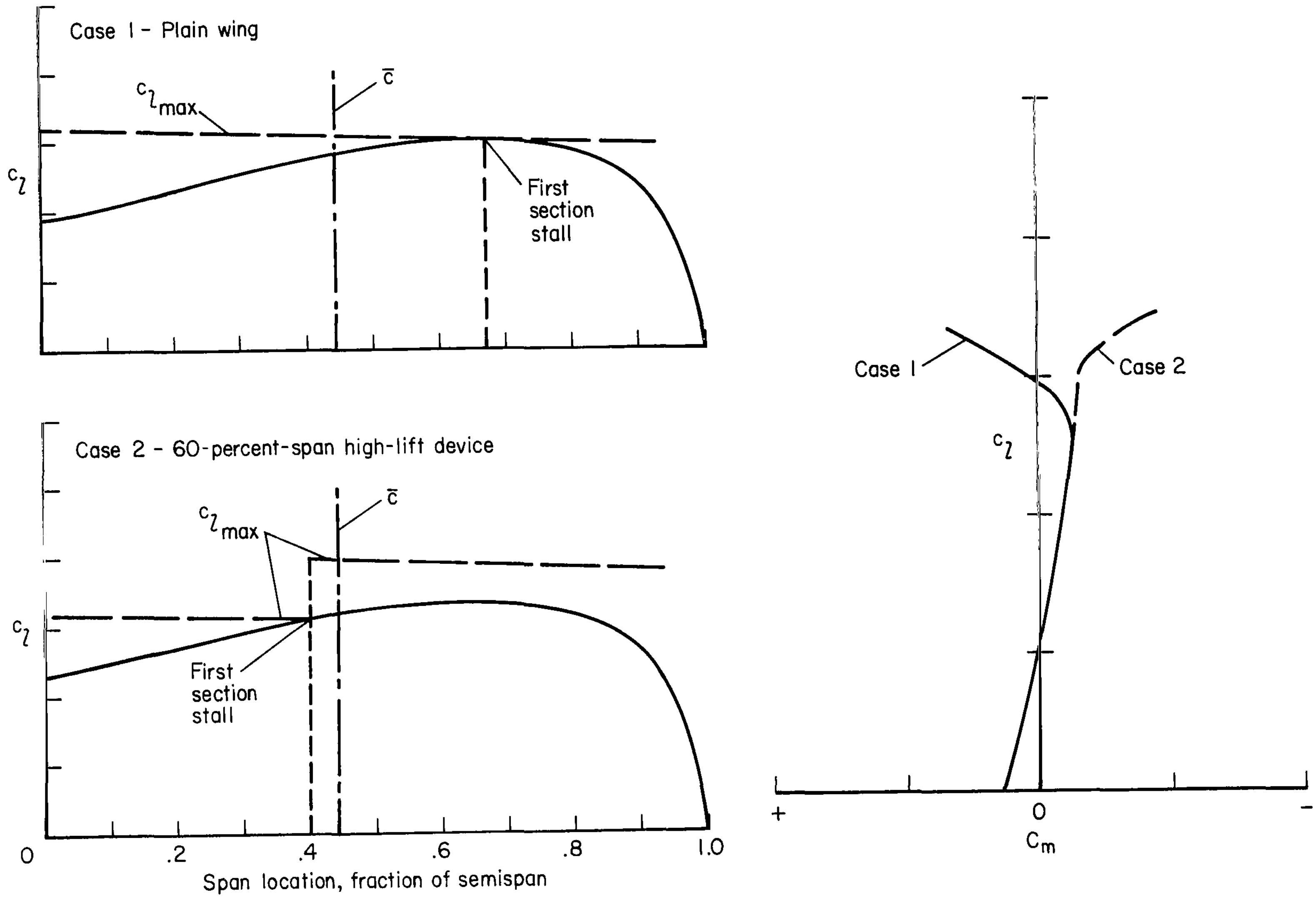


Figure 7.- The selection of high-lift devices to stabilize pitching-moment changes.

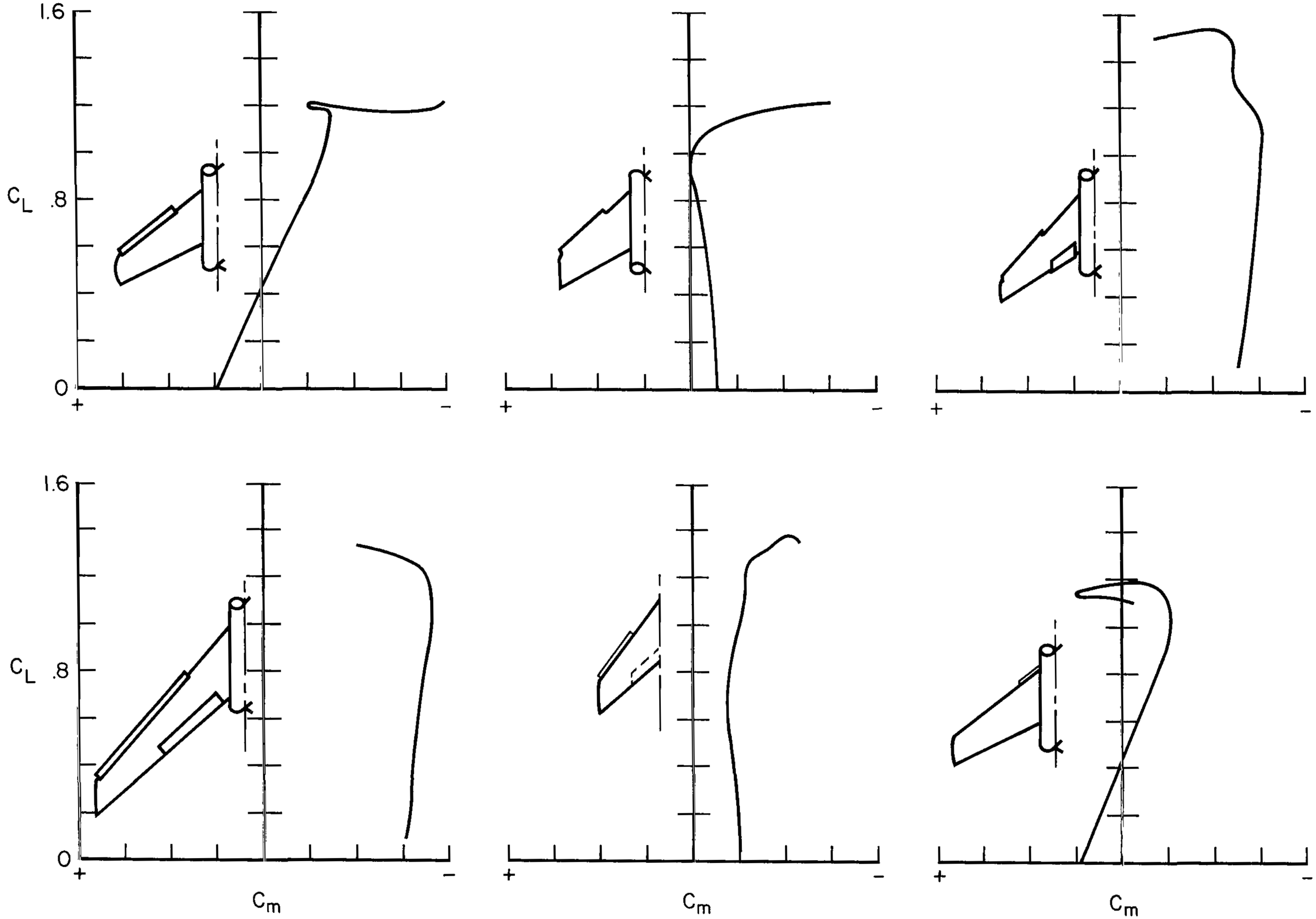


Figure 8.- Wings with high-lift devices designed for longitudinal stability at high lift.

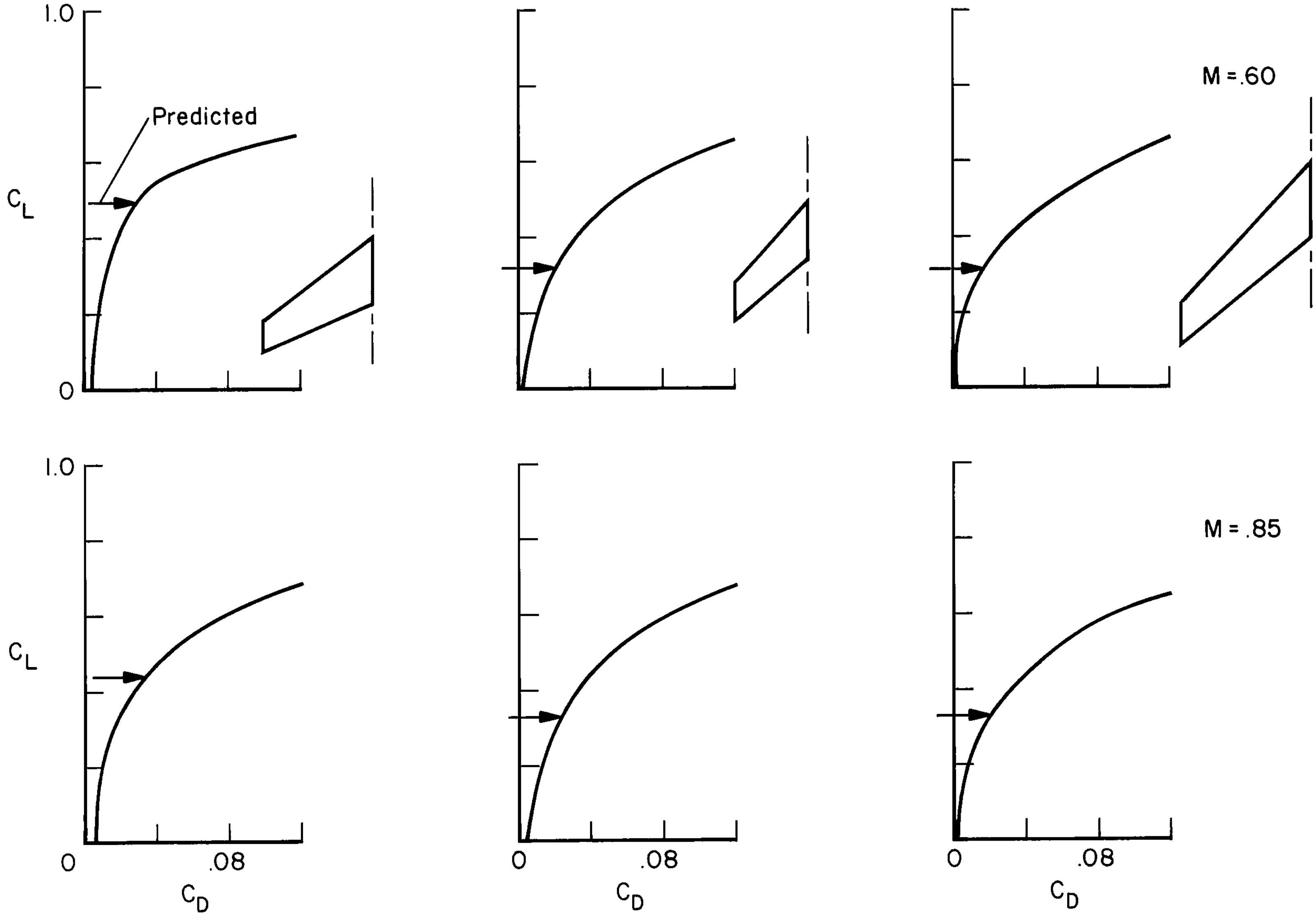


Figure 9.- Drag characteristics at high Mach numbers of several wings used to indicate first section stall.

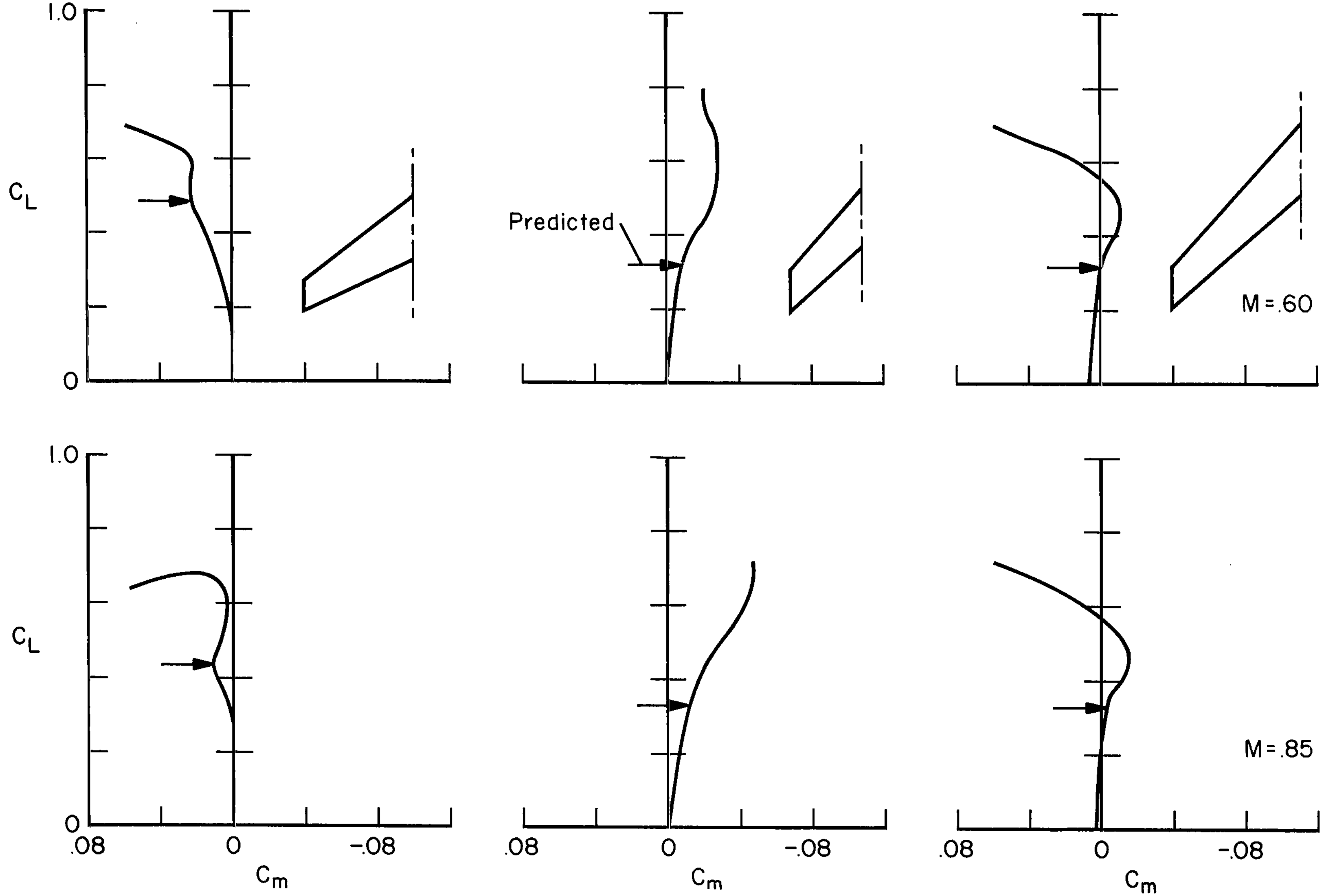


Figure 10.- Pitching-moment characteristics at high Mach numbers of several wings used to indicate first section stall.

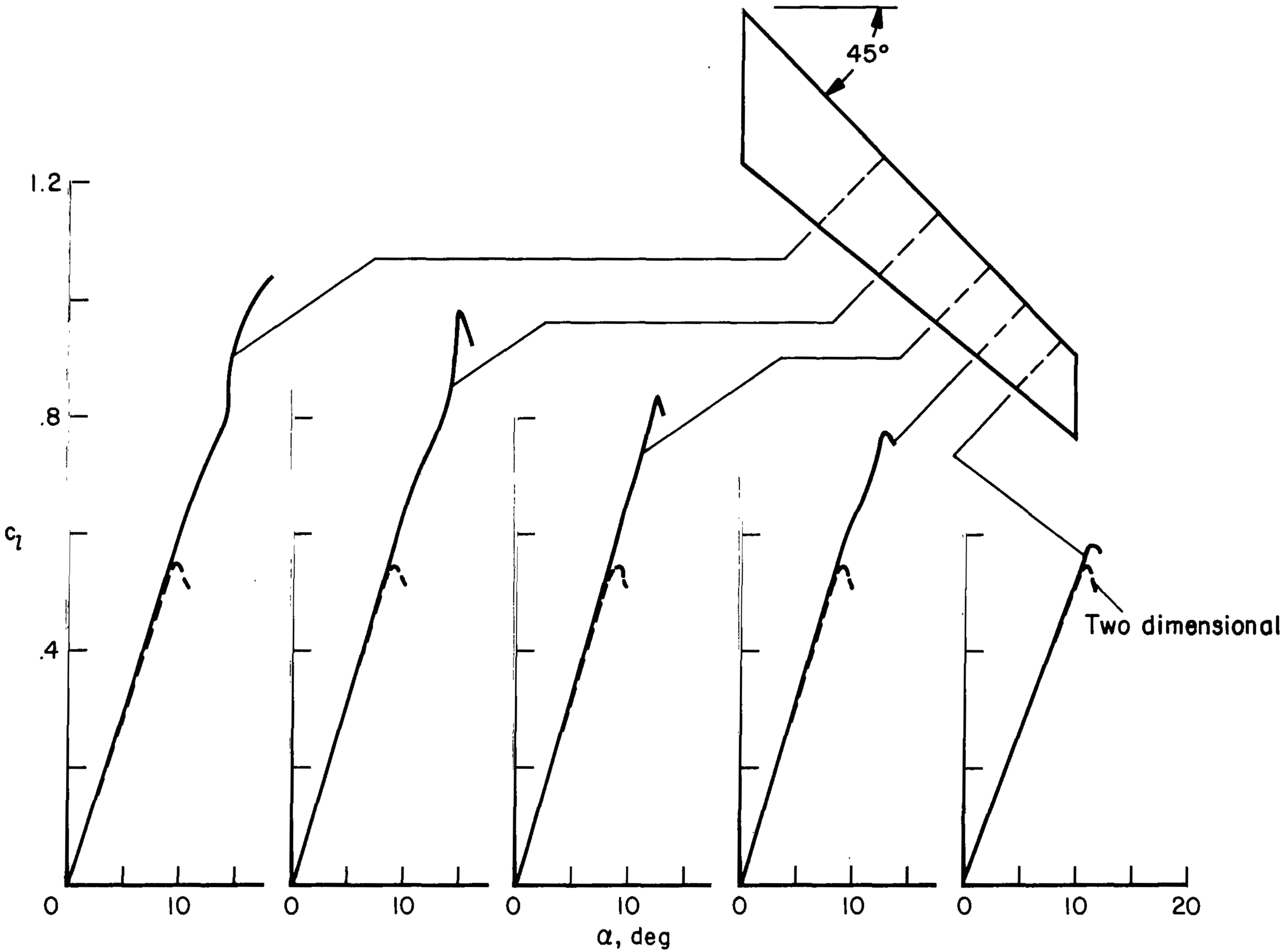


Figure 11.- Comparisons of two- and three-dimensional experimental section lift curves; NACA 64A010 sections.

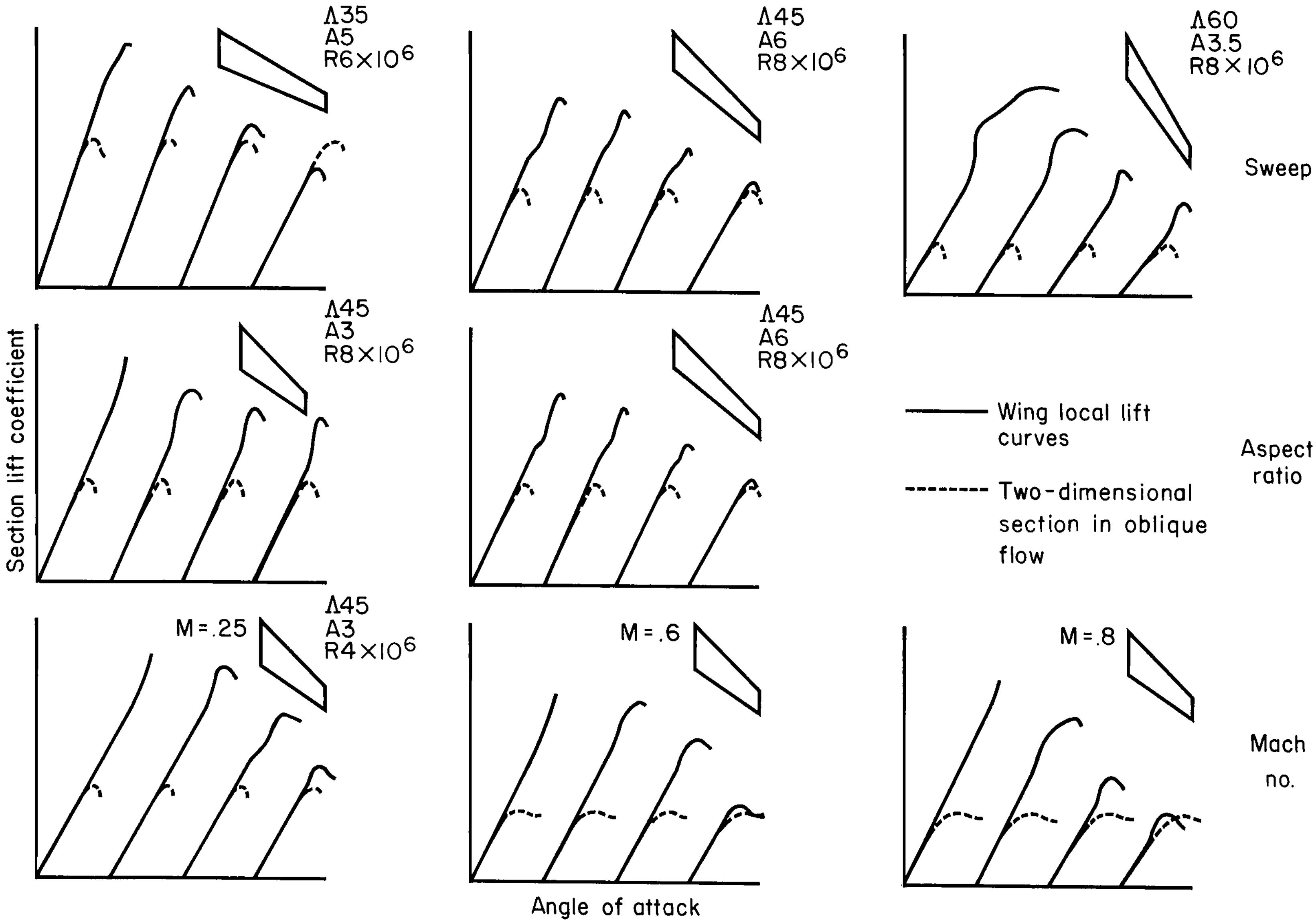


Figure 12.- Effect of sweep, aspect ratio, and Mach number on comparison of two- and three-dimensional lift curves.

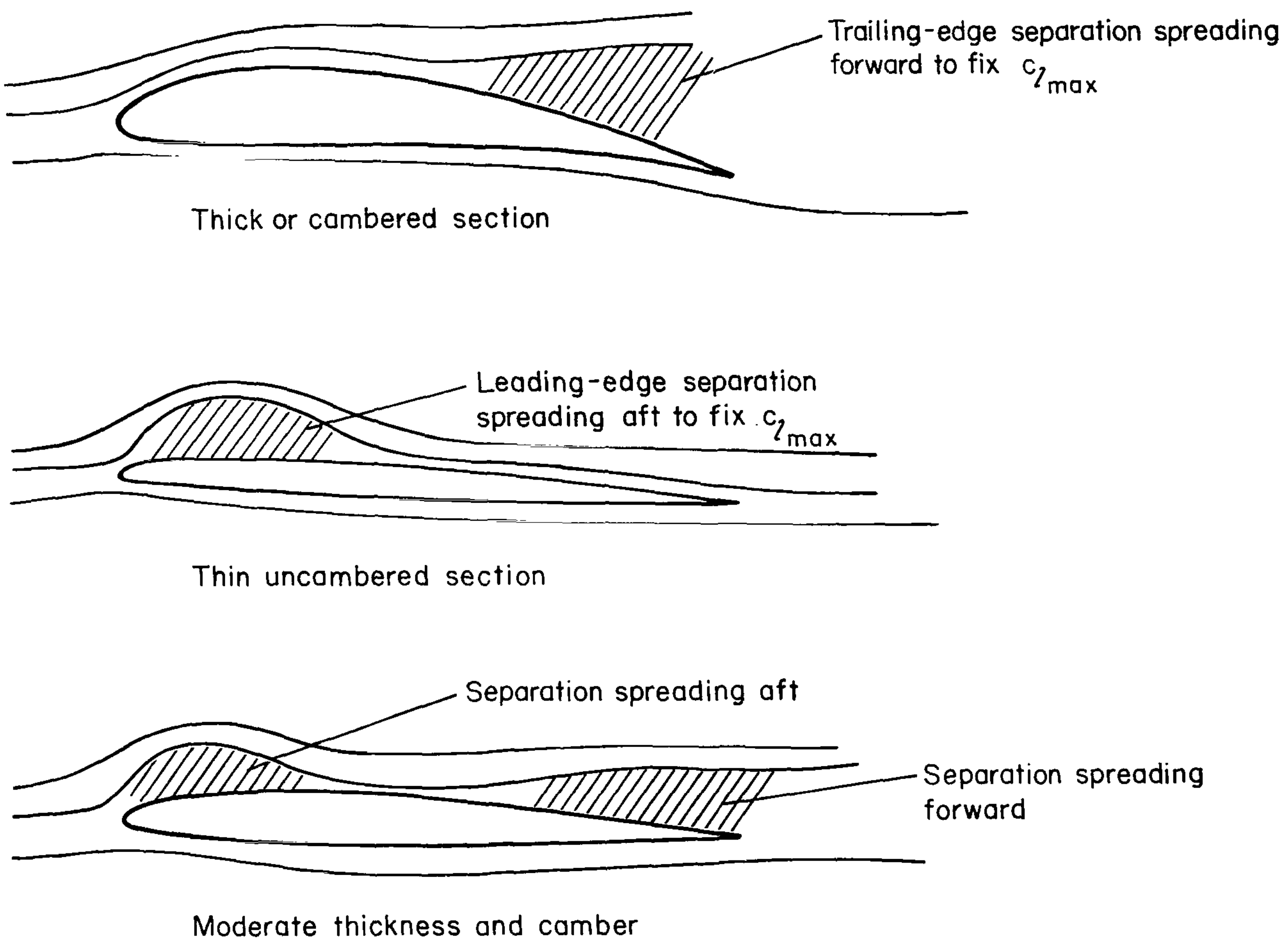


Figure 13.- Illustration of three types of section stall.

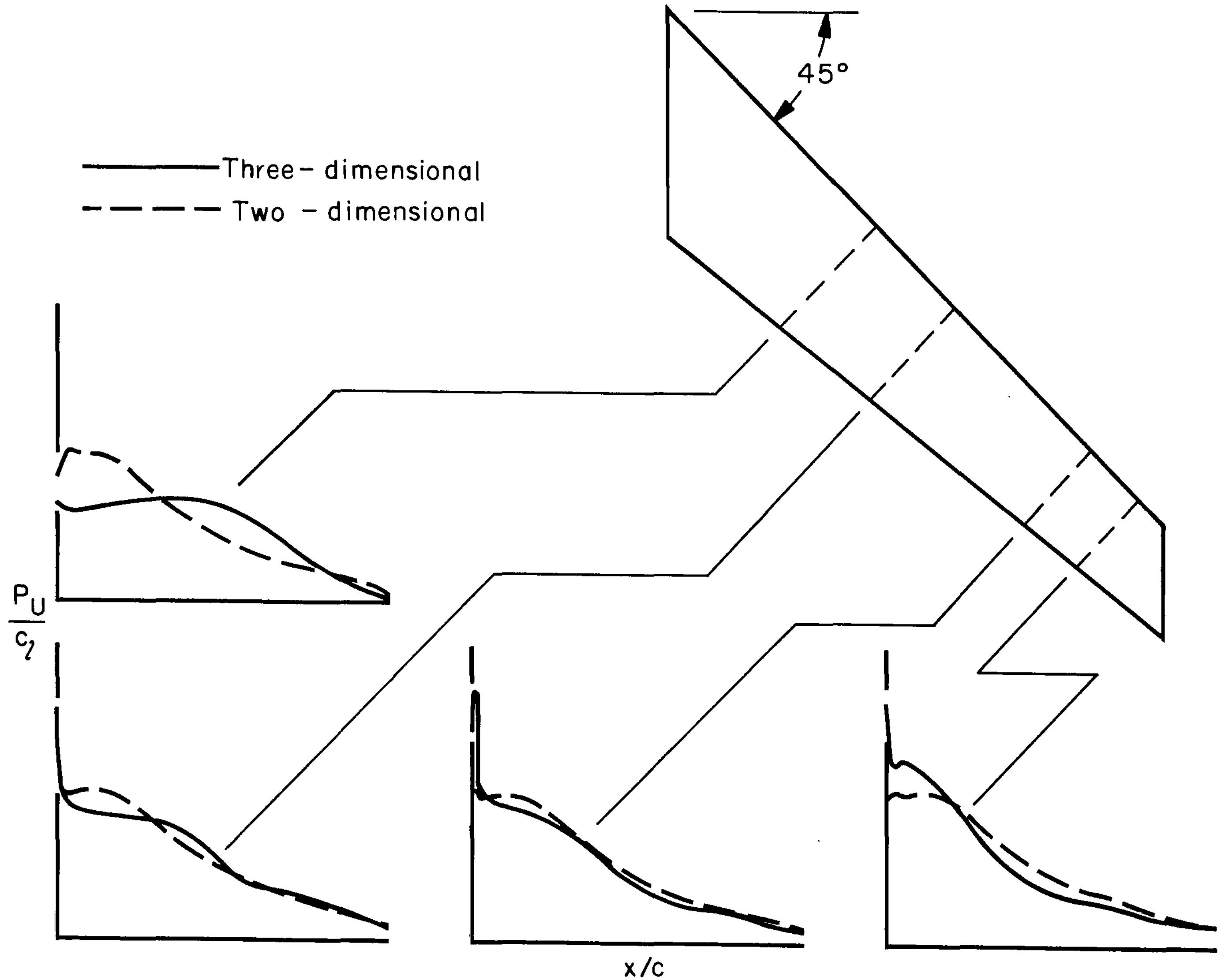
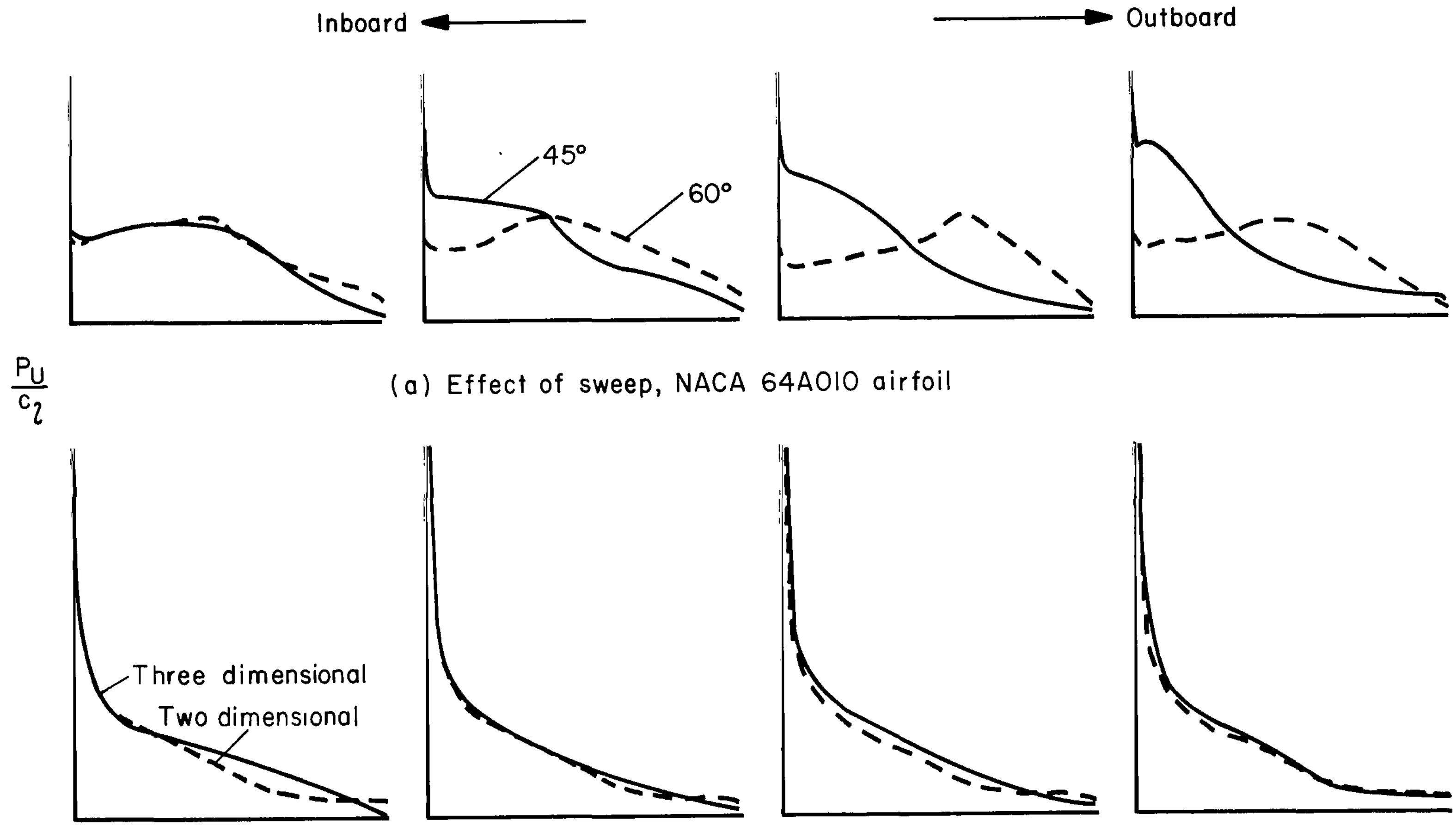


Figure 14.- Comparisons of two- and three-dimensional experimental pressure distributions at $c_{l_{max}}$; NACA 64A010 sections.



(a) Effect of sweep, NACA 64A010 airfoil

(b) Separation patterns at $c_{l_{max}}$. NACA 64A810 airfoil, 45° sweep

Figure 15.- Effect of sweep and airfoil section on separation pattern at $c_{l_{max}}$.

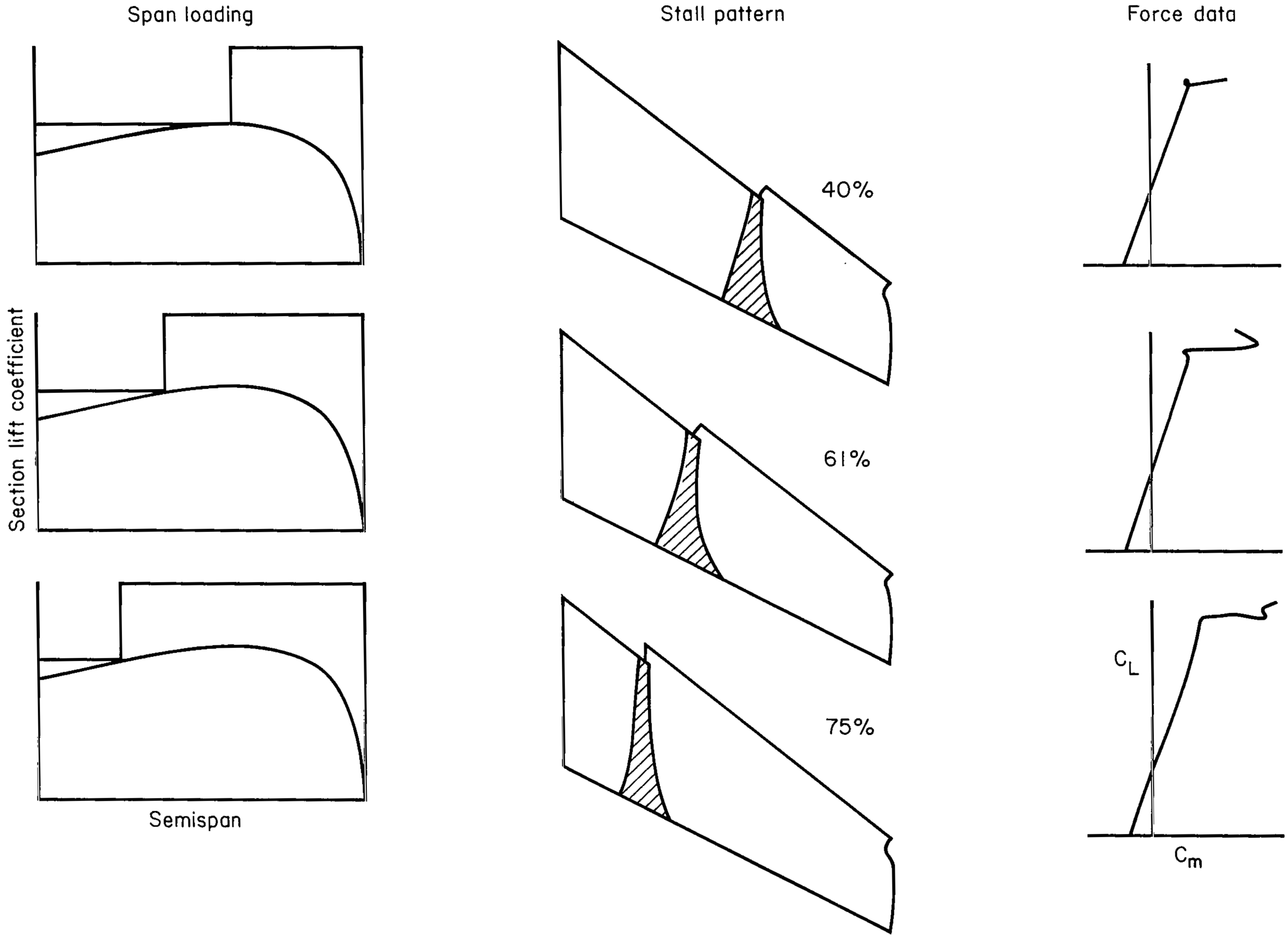


Figure 16.- Control of pitching moments by spanwise location of first stall on a wing swept 35°.

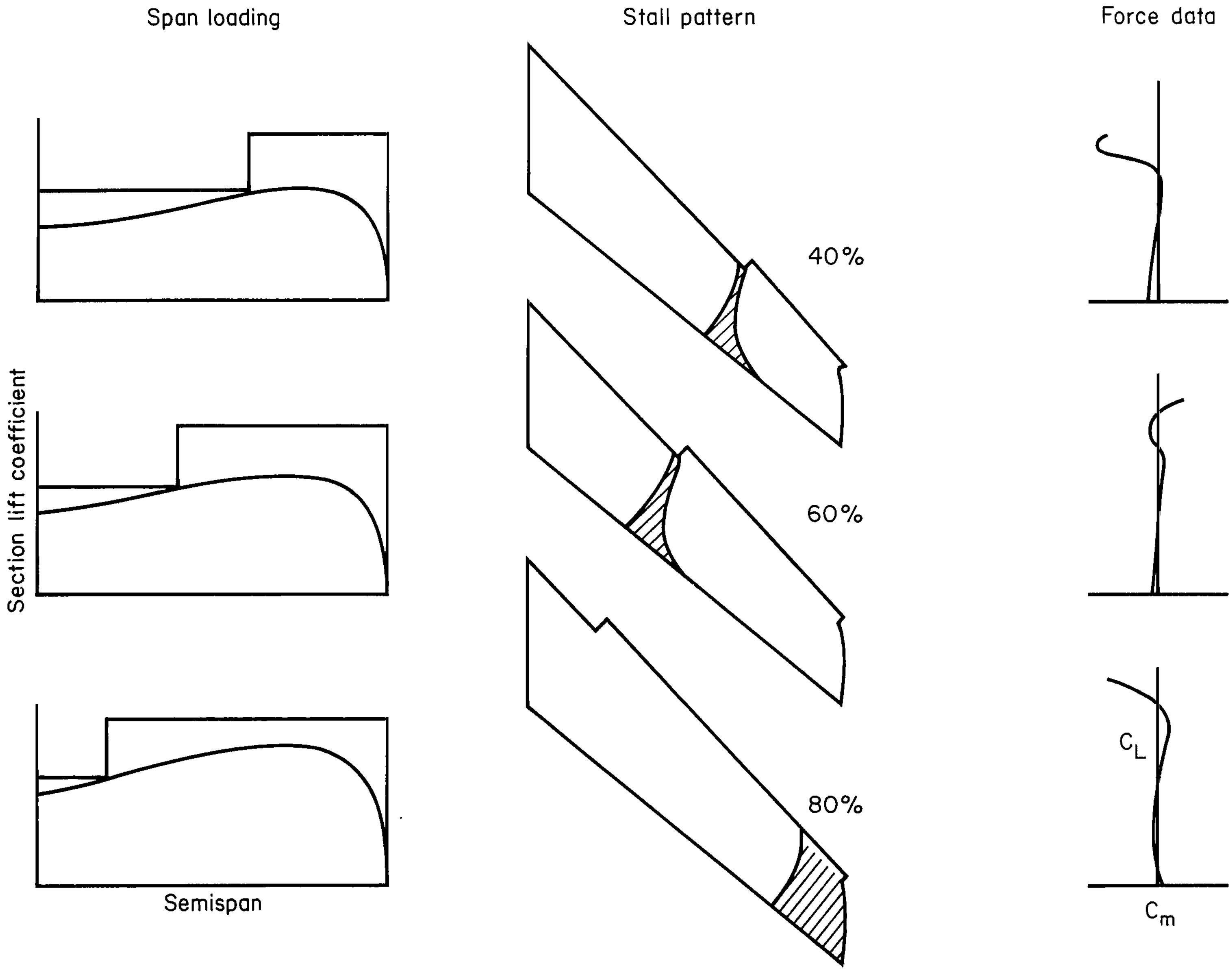


Figure 17.- Control of pitching moments by spanwise location of first stall on a wing swept 45°.

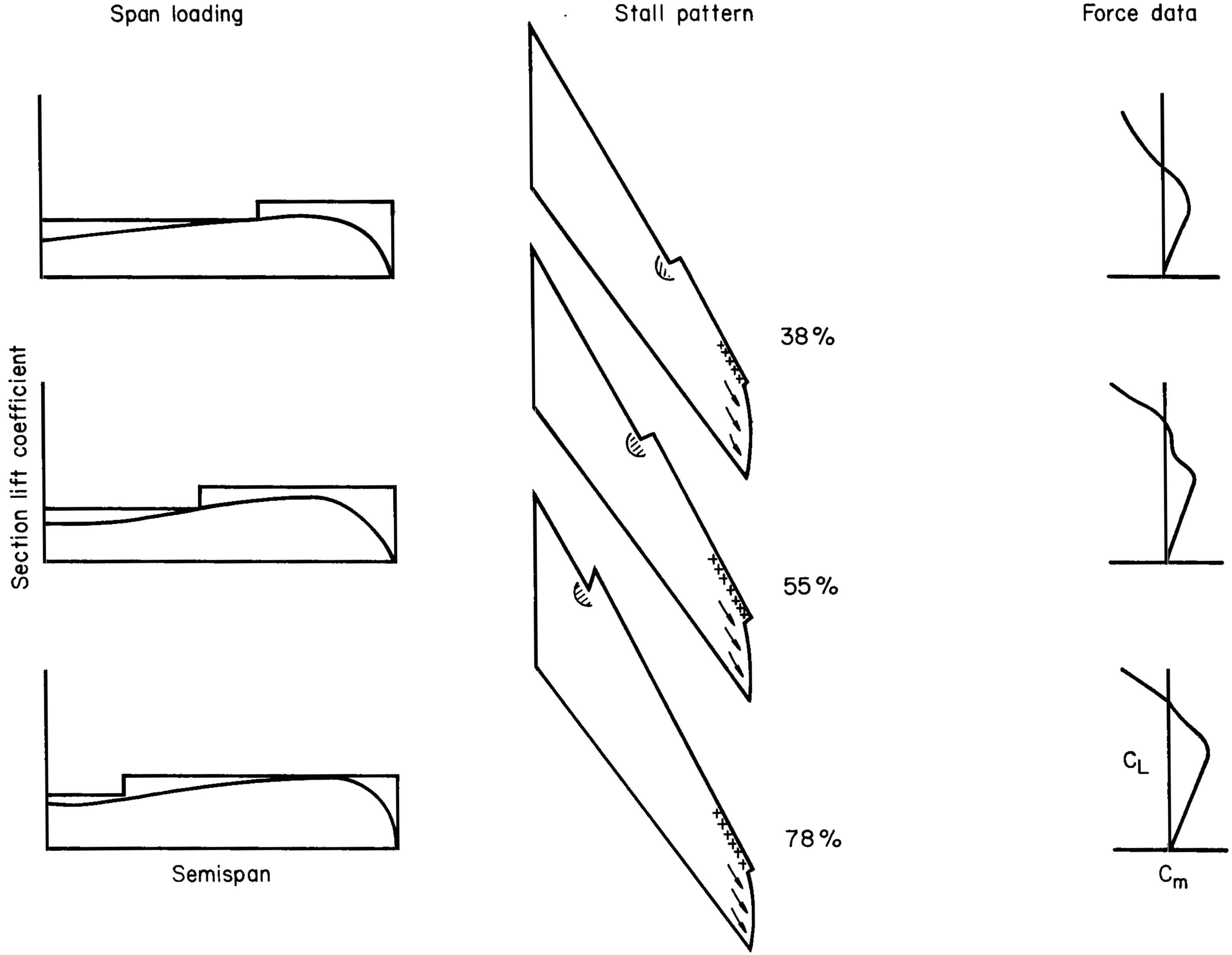


Figure 18.- Control of pitching moments by spanwise location of first stall on a wing swept 60°.

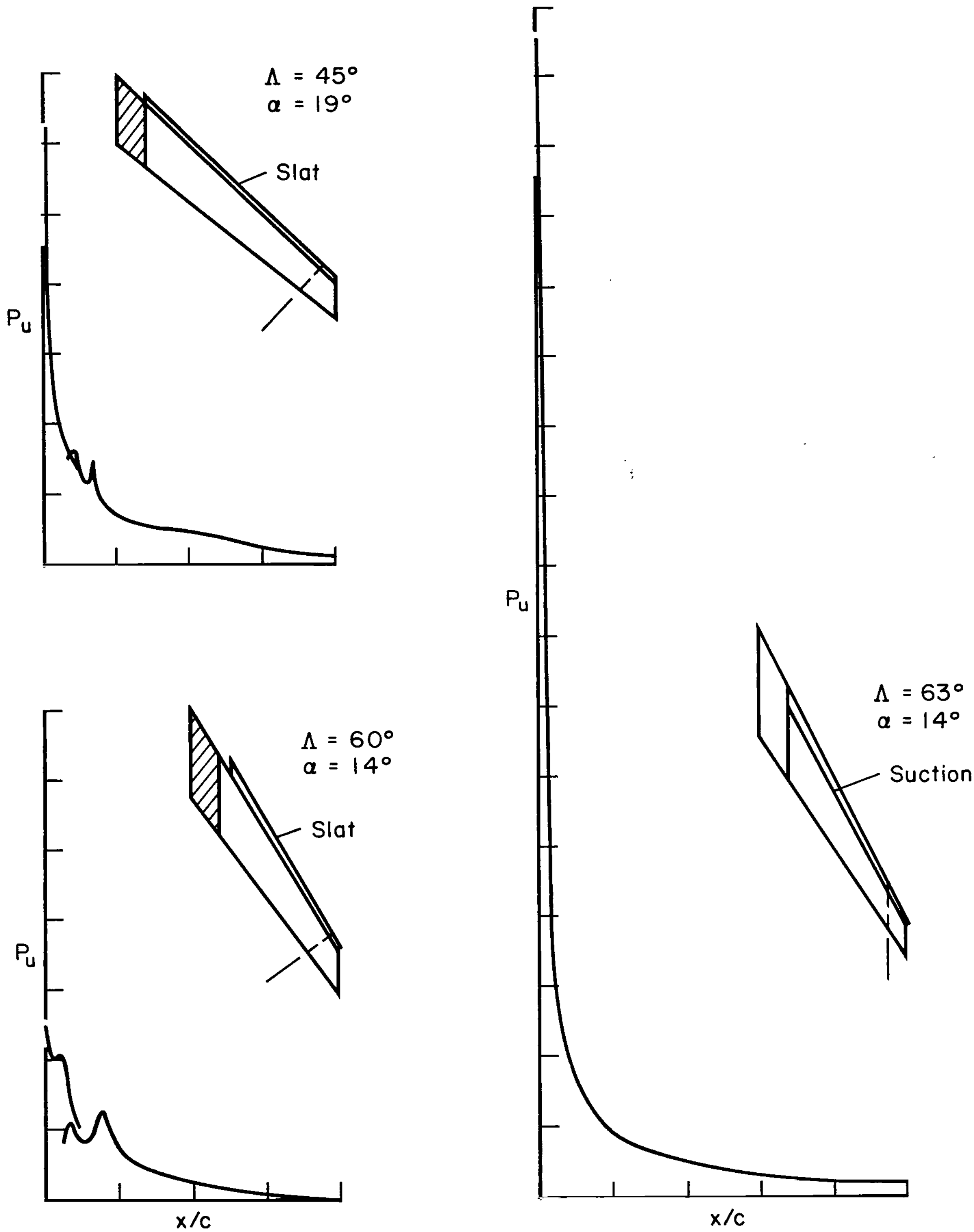
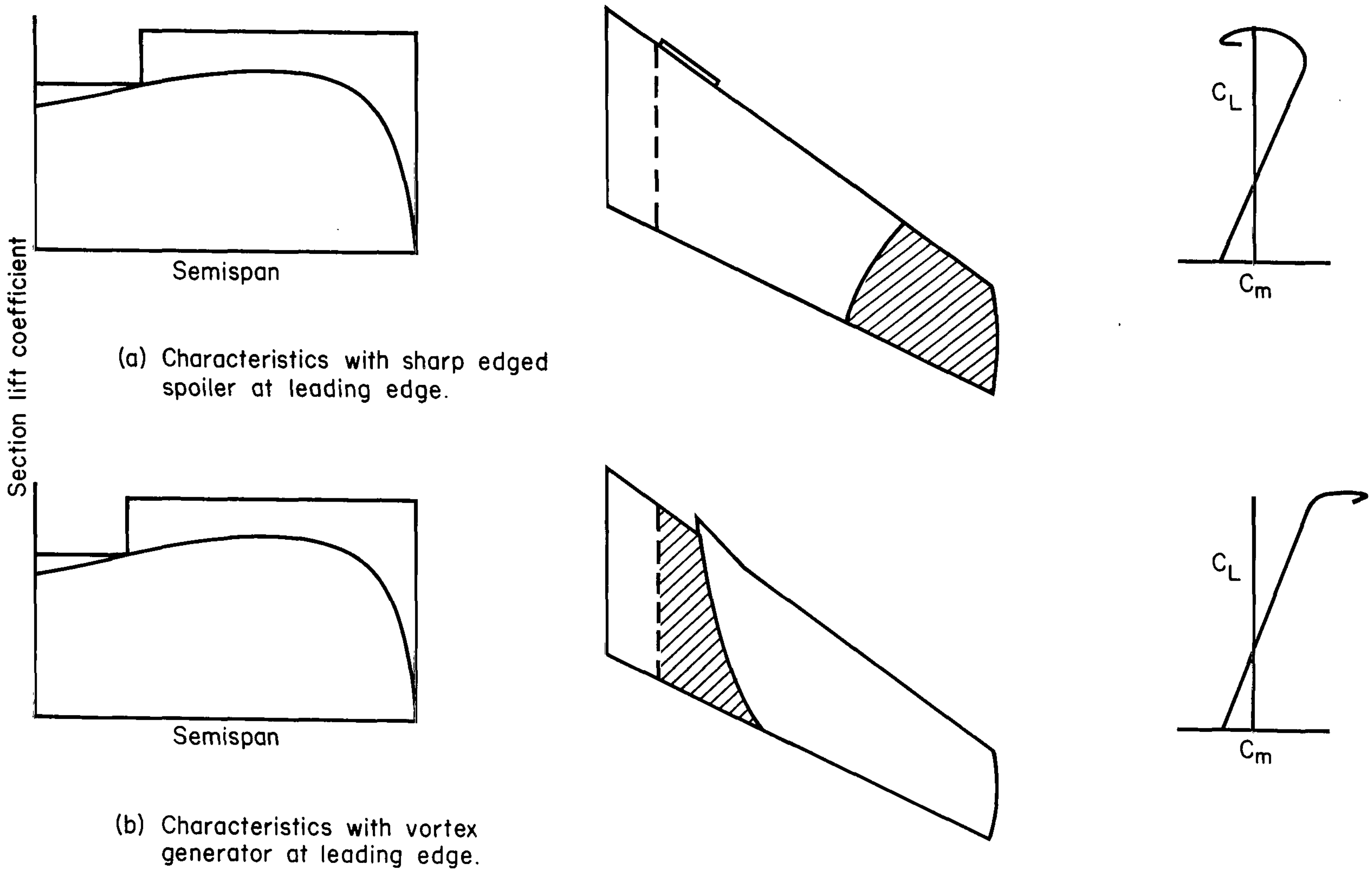


Figure 19.- Full-span stall-control devices on wings of various sweeps.



(a) Characteristics with sharp edged spoiler at leading edge.

(b) Characteristics with vortex generator at leading edge.

Figure 20.- Control of stall location by preventing boundary-layer control on inboard station; F-86 airplane.

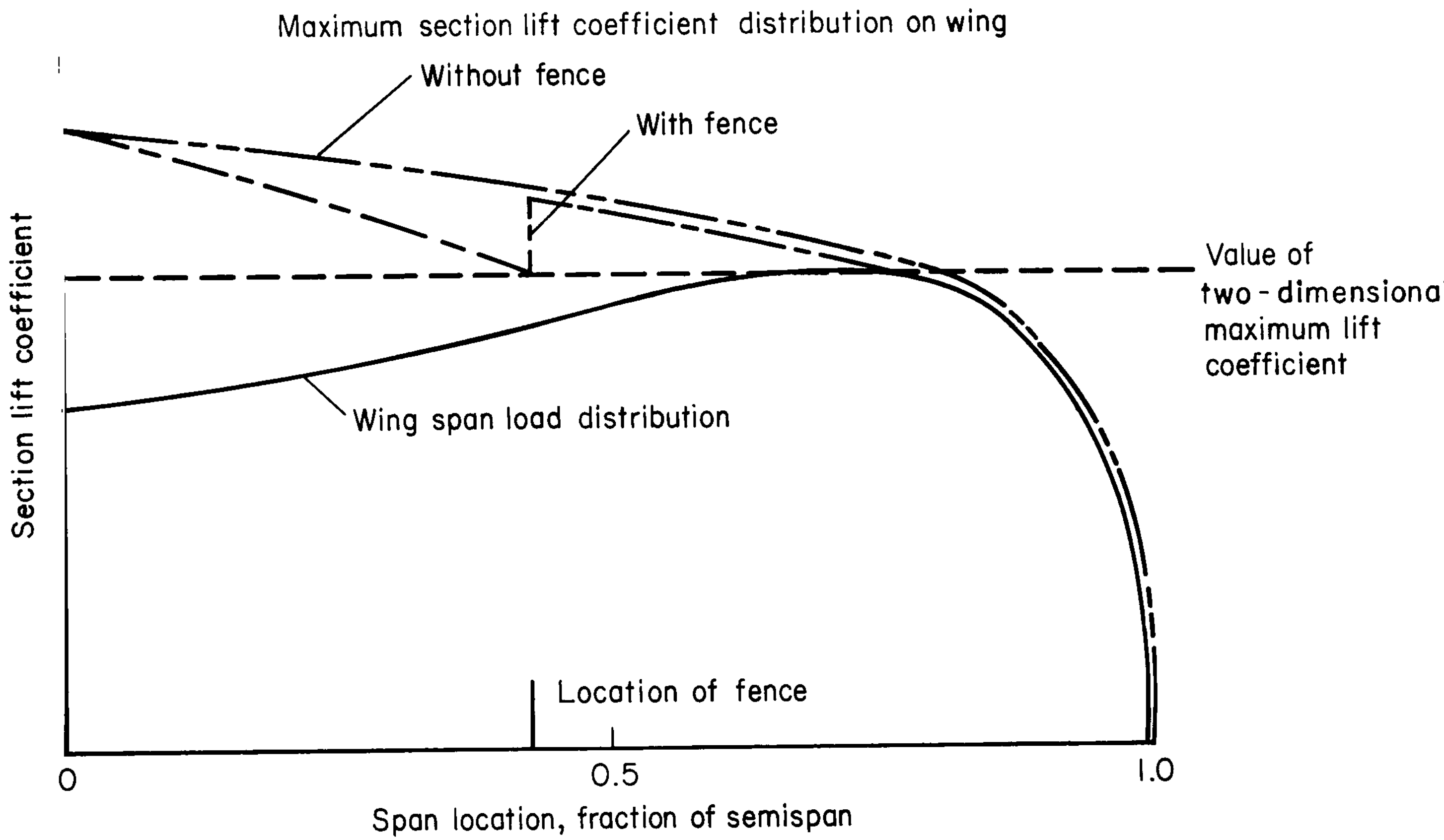


Figure 21.- The effect of a fence on the maximum lift potential across the span of a swept wing.

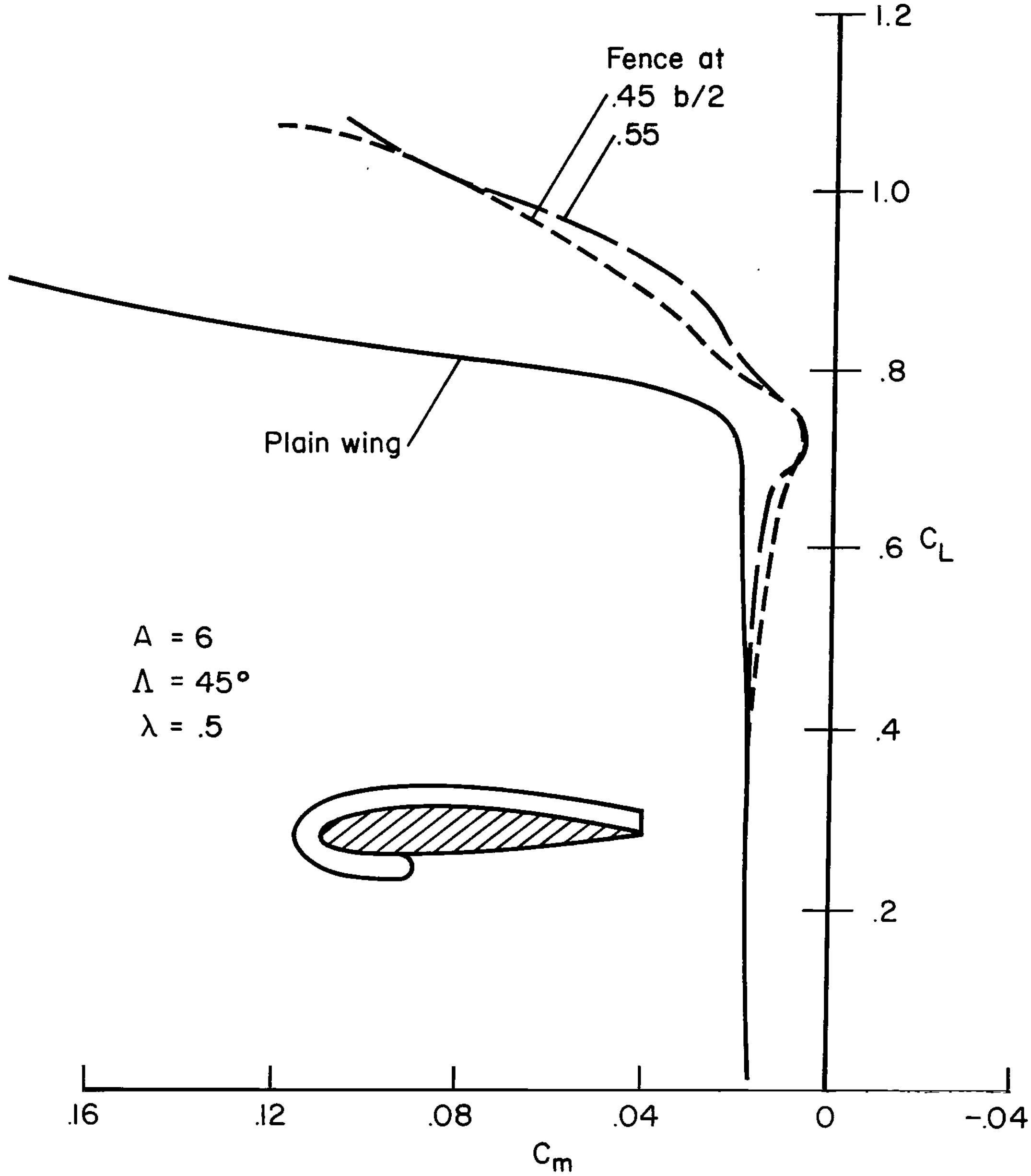
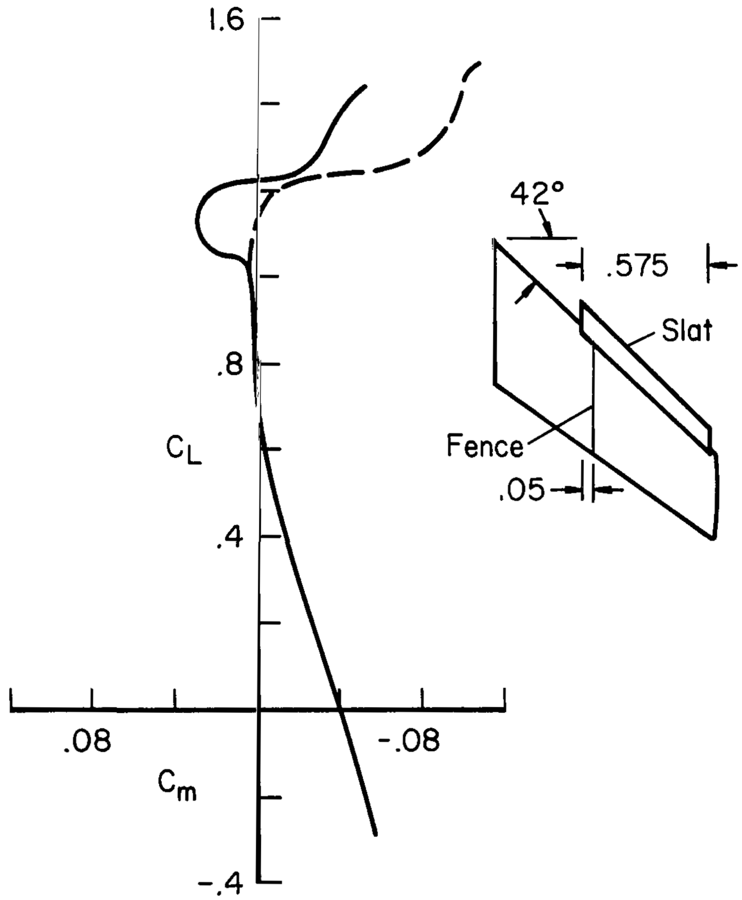


Figure 22.- Effect of fences on the pitching-moment characteristics of a swept wing.



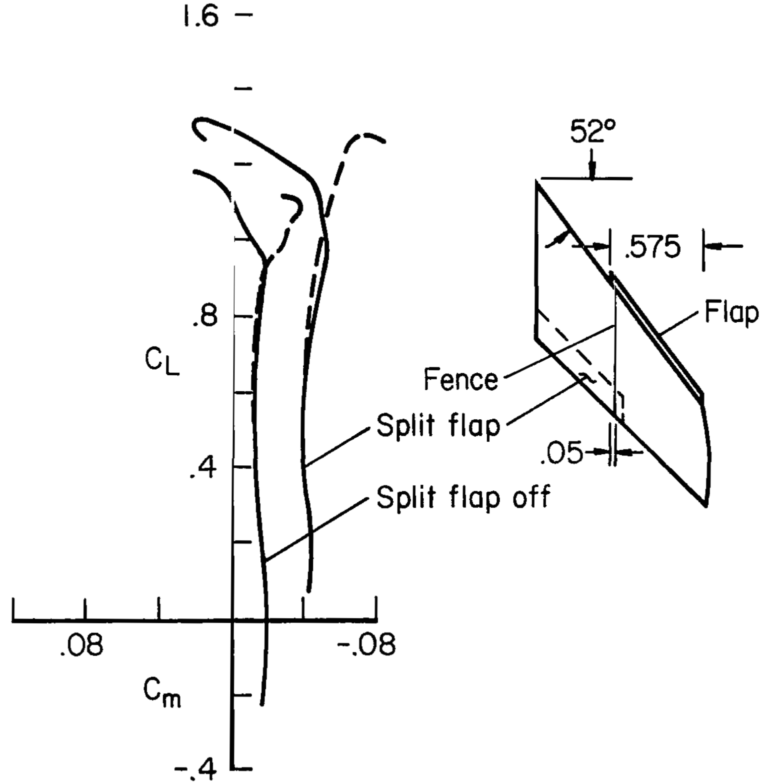


Figure 23.- Effects on pitching moments of upper surface fences in combination with partial-span leading-edge devices.

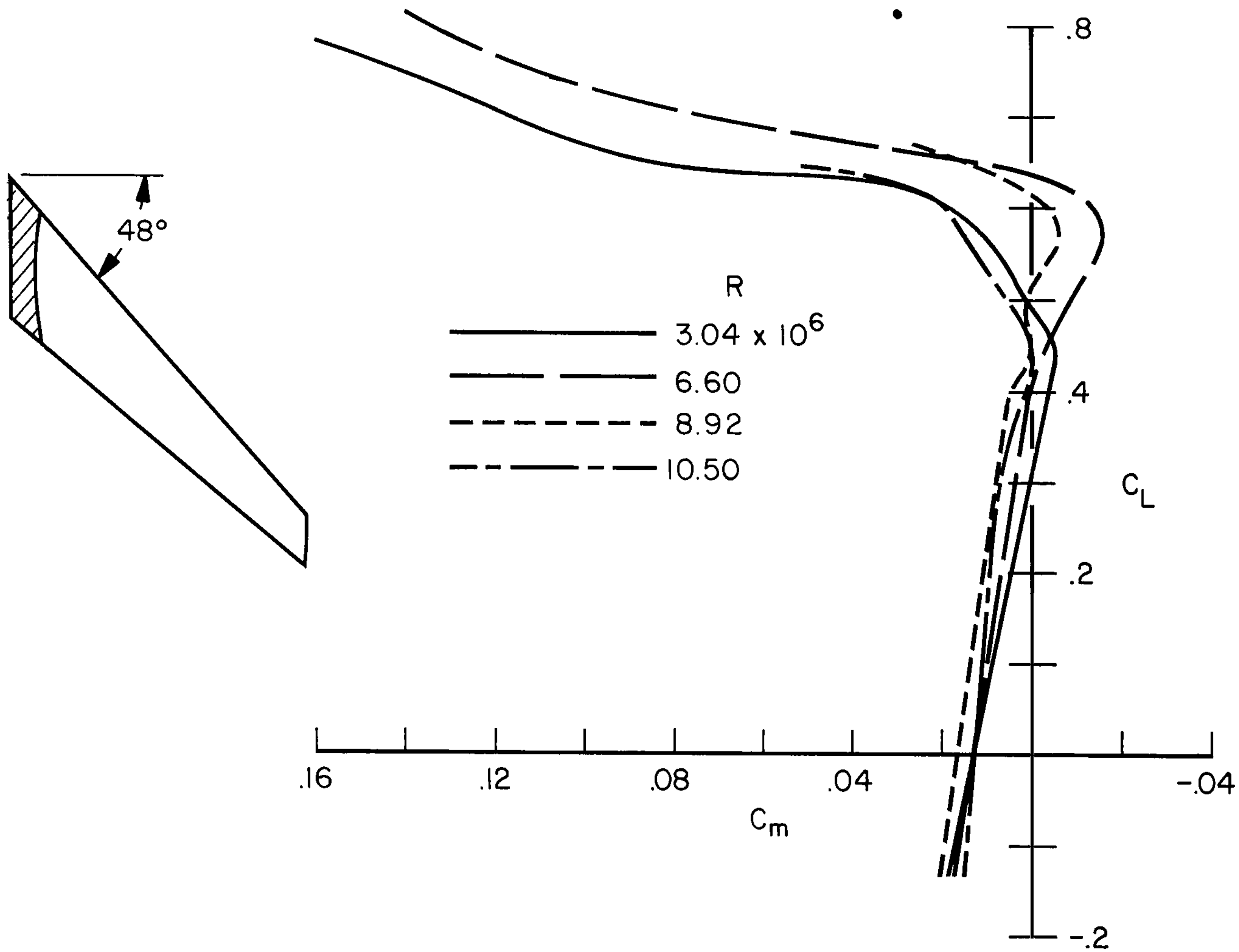


Figure 24.- Effects of Reynolds number on pitching-moment characteristics.

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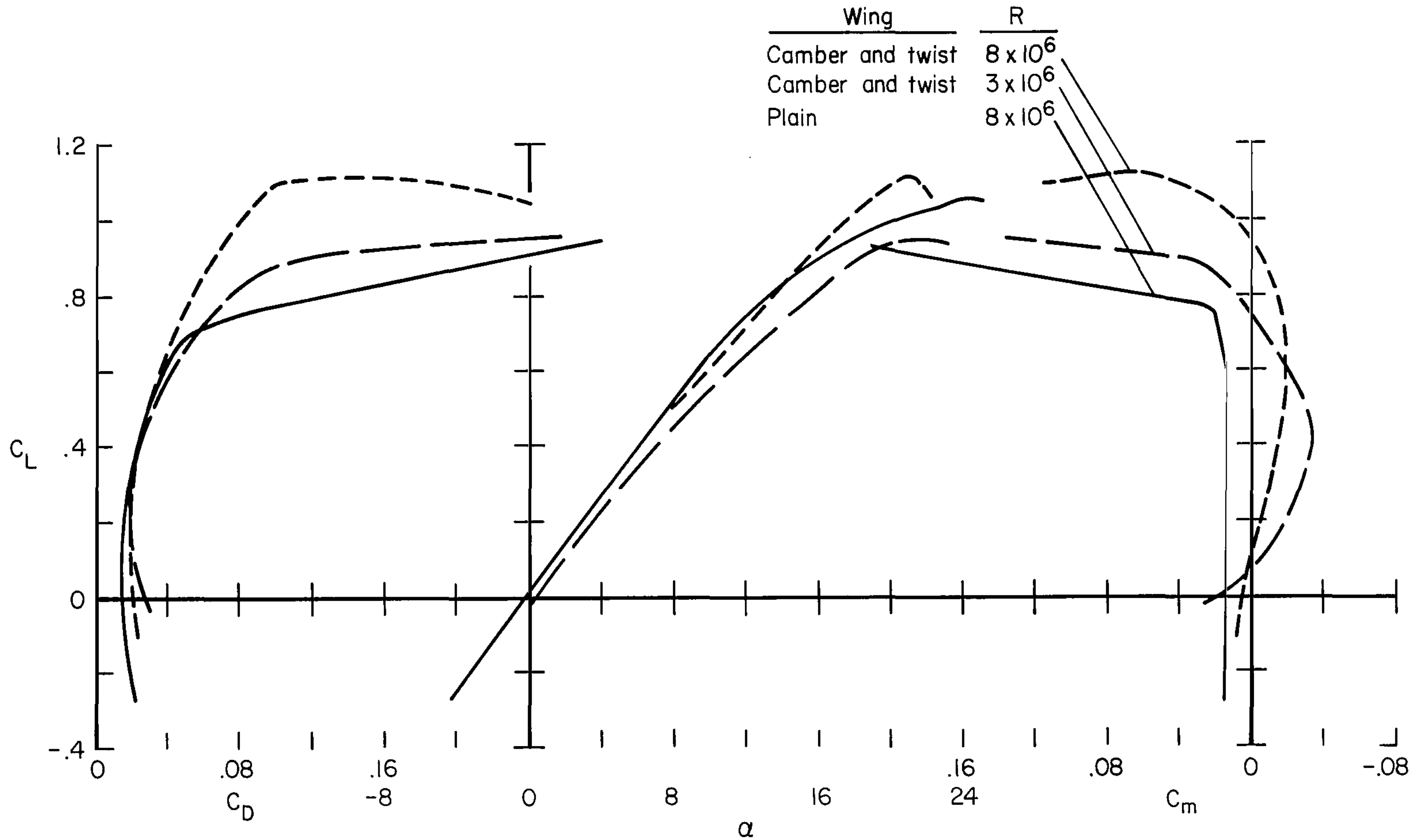

Figure 25.- Influence of Reynolds number on the effectiveness of twist and camber on a 45° swept wing.

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